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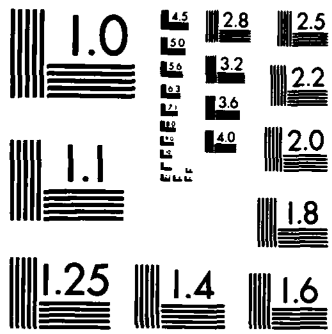

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A COMPARISON OF THE NAVTOLAND SH-2F HELICOPTER MODEL WITH THE REQUIREMENTS OF MIL-H-8501A AND WITH FLIGHT DATA

AD-A157 297

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This report uses an SH-2F Simulation Model to examine the significance of some of the quantitative requirements of MIL-H-8501A. Performance of the model is calculated and compared with both flight test data and the requirements of MIL-H-8501A.

Additions and Modifications to the current flying qualities specification are recommended.

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SUMMARY

This report compares the dynamic response characteristics of an SH-2F simulator model with available flight test data and with the requirements of MIL-H-8501A, Helicopter Flying and Ground Handling Qualities; General Requirements for. The Navy Vertical Takeoff and Landing (NAVTOLAND) simulator model was developed to facilitate the design of an automatic approach and landing guidance control system for helicopters and V/STOL aircraft. The simulation was validated by comparing the model response with all available flight test data. In addition, the model was used to examine some of the requirements of MIL-H-8501A. Considerable insight was gained in the application of the specification and in understanding techniques needed to determine compliance with the requirements.

The original model was developed and installed on the NASA Ames Vertical Motion Simulator. ~~(VMS)~~ Numerous modifications were made to the software based on test pilot evaluations. A version of this simulation model including all empirical adjustments was installed on the NAVAIRDEVCEN computer system. Further evaluation of the model was carried out using this batch version of the helicopter model. This study found

Experience gained in this study showed that it is difficult if not impossible to satisfactorily tune a helicopter simulation based on pilot opinions without a complete flight test data base.

The requirements of MIL-H-8501A need to be defined in more detail and in some cases be revised. An altitude excursion limit should be added to the current quick stop maneuver. ~~A~~ lower bound should be placed on the time to double amplitude for aperiodic divergences in addition to the current limit on oscillatory divergences. Analysis ~~also~~ indicates that the current 2-second two second pilot response delay following engine failure may be excessive.

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LIST OF SYMBOLS

Symbol	
ASE	Automatic Stabilization Equipment
C3	Aircraft vertical velocity - +up, ft/sec
$I_{X, Y, Z}$	Aircraft inertia components - slug-ft^2
L_p	Roll damping - rad/sec^2/inch
L_V	Static Dihedral Rolling Moment Derivative - $\frac{\text{rad/sec}^2}{\text{ft/sec}}$
L_δ	Roll control power derivative - rad/sec^2/inch
M_δ	Pitch control power derivative - rad/sec^2/inch
NAVTOLAND	Navy Vertical Takeoff and Landing System
N_V	Static Directional Stability Derivative - $\frac{\text{rad/sec}^2}{\text{ft/sec}}$
P	Aircraft roll rate - deg/sec
PHIDeg	Bank angle - deg
PSIDeg	Yaw angle - deg
Q	Aircraft Pitch Rate - deg/sec
R	Aircraft Yaw Rate - deg/sec
R/C	Rate of Climb - ft/minute
RPM	Rotor speed - revolution/minute
THETDeg	Pitch angle - deg
U	Longitudinal airspeed component - ft/sec
U_c	Commanded longitudinal velocity - ft/sec
U gust, V gust, W gust,	Simulated gust velocity components - ft/sec

data is needed to establish minimum acceptable boundaries for longitudinal speed stability. It is likely that the minimum acceptable boundaries will depend on the maneuver task as well as the available visual cues. However, the overall minimum requirement should be determined by the most critical task for the given helicopter.

Figure 6 compares the model trim with flight test data for forward speeds from 40 to 130 knots. The model shows excellent correlation with the flight test data in pitch attitude between 40 knots and 120 knots. Above 120 knots the model predicts excessive nose down attitude. Longitudinal stick matches well up to 100 knots. Lateral stick shows the greatest mismatch at low speeds with the error changing sign and increasing again above 120 knots. The rudder pedal position shows the proper trend with speed, but has the greatest discrepancy at 40 knots.

Both the model and the flight test data show a vary slightly stable longitudinal gradient above 40 knots. The gradient becomes more stable above 100 knots with the model somewhat overestimating the amount of forward stick required to trim.

However, this comparison of level flight trim characteristics does not precisely define the speed stability characteristics as implied in Section 3.2.10 of MIL-H-8501A. Three alternate approaches for determining speed stability were examined using the SH-2F analytic model. In the first approach, the model was trimmed in level flight at selected speeds. Horizontal speed was then perturbed with all controls fixed. Figure 7 illustrates the variation of pitching moment and horizontal force versus speed with vertical velocity fixed at zero and all controls set to the local trim values. This analysis technique gives a good indication of longitudinal gust stability in that the computed perturbations correspond to a sudden change in the horizontal relative wind with no change in any other state variable. Figure 7 shows a trend similar to figure 6. The pitching moment gradient is stable everywhere but has a very small positive value near 60 knots air speed. Horizontal speed damping increases monotonically with airspeed. Equivalent stick perturbation was computed by dividing the moment perturbation by the longitudinal stick control power to plot the data in a conventional format. Equivalent pitch attitude perturbation was also calculated as:

$$\Delta\Theta_{EQ} = \frac{\Delta \text{ longitudinal force}}{\text{weight}} \quad \text{(rad.)}$$

Following this analysis the helicopter model trim algorithm was modified to allow sink rate and longitudinal stick to vary with collective stick fixed.

The helicopter was first trimmed in level flight at a specified airspeed by allowing the pitch and bank attitude and the four control positions to vary. Horizontal speed was perturbed by ± 10 knots and the sink rate and longitudinal stick were varied with collective fixed to establish a new trim. This procedure agrees with the intent of MIL-H-8501A. However, the engine torque was varied automatically by the engine governor. Thus, the requirement for fixed throttle was not duplicated. Figure 8 illustrates the stick gradients and vertical rates of climb produced by the model analysis. The stick gradients calculated in figures 8 and 9 are nearly identical. This suggests that forces and moments caused by changes in vertical rate have minimal effect on the total pitching moment and the resultant longitudinal stick position.

In an alternate approach, longitudinal speed stability was investigated by calculating dynamic time responses. The ASE model was used to approximate piloted control using the ground speed hold mode with fixed collective control. Both forward and aft airspeed change commands were introduced for a range of trim level flight speeds. Speed stability was established by calculating

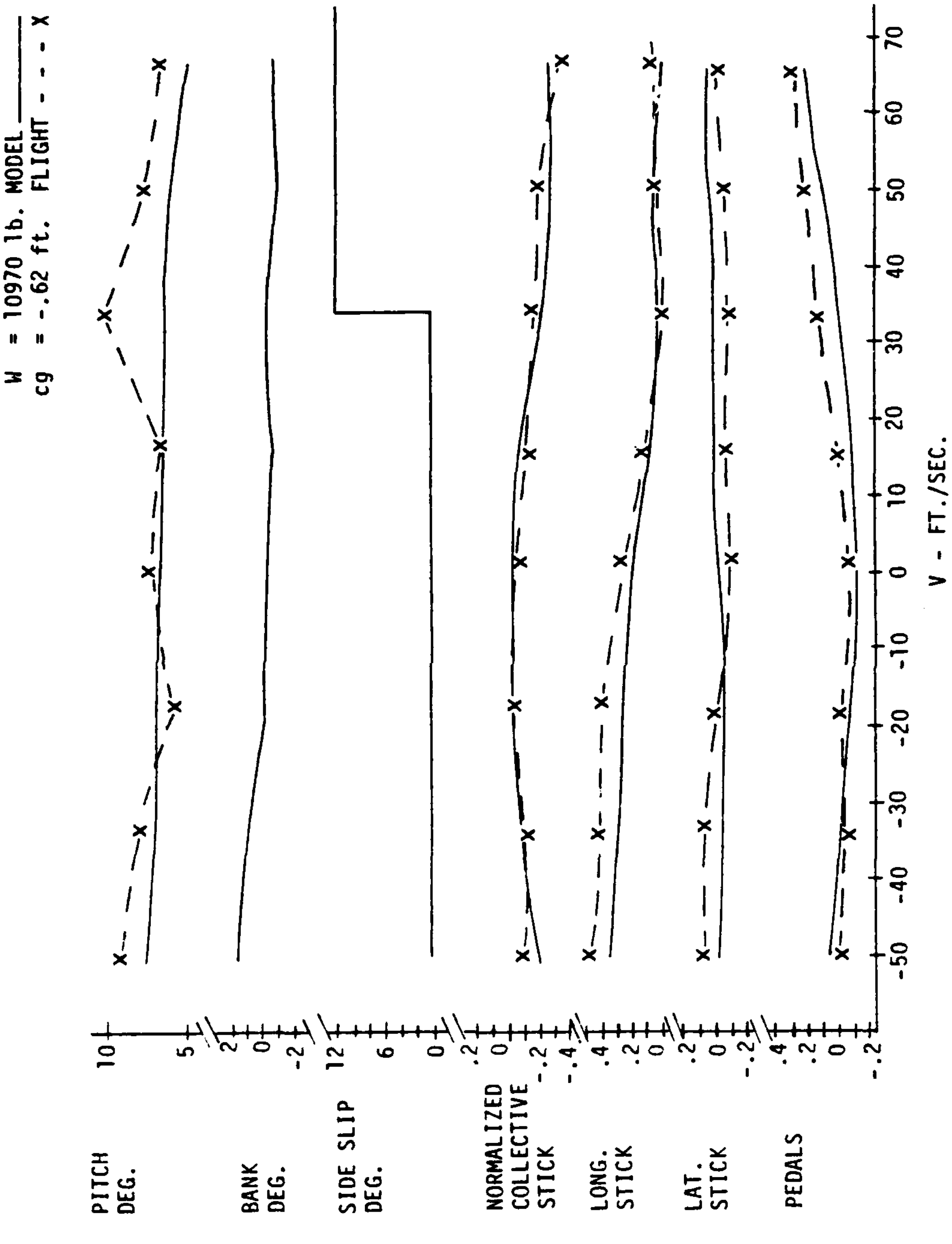

Figure 5. SH-2F Model Low Speed Trim Characteristics vs. Flight Test

The pilot uses stick force and stick position as a strong kinesthetic cue for stabilizing the aircraft. In general, increasing the pitch attitude reduces airspeed. Aft stick motion should always be required to increase pitch attitude and forward stick must always produce decreased pitch attitude because the pitch attitude closure is the most basic and the highest frequency task performed by the pilot in longitudinal control. Thus, no perceptible degree of control reversal can be tolerated in the moment generated.

Airspeed is regulated at a lower frequency than pitch attitude. However, speed regulation can be complicated by the non-linear trim characteristics of the helicopter. The SH-2F trim pitch attitude increases with forward speed from 10 knots aft to 20 knots forward speed and then decreases at higher speeds. Thus, there is a range of speeds where a given value of pitch attitude corresponds to three different trim airspeeds. Therefore, pitch attitude can be an unreliable reference for controlling speed near hover. The low speed flying qualities of the SH-2F would be undoubtedly improved if a monotonic relationship existed between pitch attitude and airspeed for all flight conditions.

Data obtained from reference (e) indicates that both the AH-1G and UH-1H have unstable pitch attitude versus airspeed gradients in rearward or low forward speed flights. The OH-6A and BO-105C have monotonically stable pitch attitude versus speed gradients. Unfortunately, pilot opinion data needed to confirm the significance of the pitch attitude gradient versus speed is either limited or unavailable. However, various manufacturers had difficulty in certifying their helicopters for decelerating IFR approaches to a stabilized hover. Unstable pitch versus speed gradients may be a factor in these difficulties.

A certain degree of control position reversal with forward trim speed can be tolerated as long as the stick force gradient requires increased push for increased forward speed. A stable stick force gradient tends to mask stick position instability because the pilot senses force more directly than position.

Longitudinal stick speed stability was examined by calculating the trim longitudinal stick position of the SH-2F model for a range of level flight speeds. Collective control position was allowed to vary as required to achieve trim. In order to exactly satisfy the conditions of the specification, the trim calculations were repeated with the collective stick fixed and the sink rate and longitudinal stick allowed to vary as the speed was perturbed about an initial level flight trim condition.

The specification requires that both collective and throttle controls be held fixed. This implies constant engine torque. In most modern helicopters, the engine governor maintains rotor speed by varying the drive torque as required. Thus, it is not possible to exactly duplicate the provisions of the specification unless the speed governor is disengaged. This section of the specification should be rewritten to require constant throttle setting only if the helicopter does not have a speed governor.

Longitudinal trim characteristics of the SH-2F model were compared with flight test data obtained from reference (b). Figure 5 summarizes this comparison for a range of speeds from 30 knots aft to 40 knots forward. All control positions of the model agree closely with the flight test data. However, model pitch attitude has significant mismatch near hover. This error may be caused by variation in the rotor downwash acting on the tail which is not fully accounted for by the model. The overall longitudinal stick gradient is stable from -30 to +40 knots for both the model and the aircraft. However, the flight test data indicates a slight control reversal between 20 and 40 knots. The indicated magnitude of the reversal is about ½ inch based on a longitudinal throw of ± 7 inches. This just meets the requirements of MIL-H-8501A. Additional pilot opinion position and the collective stick comes within ½ inch of full down. Rudder pedal motion is well within limits and shows very smooth gradual motion. Altitude exceeds the commanded value during the deceleration because of the limit in collective authority, but it returns to the desired altitude as the helicopter slows to hover. Based on the simulated quick-stop, an average deceleration of 10 ft/sec^2 can be achieved by the model.

Comparison with available flight test data for the AH-1T and CH-53E helicopters show that accelerations achieved in flight are typically lower than this. Test data for the AH-1T found in reference (c) show a deceleration of 3 ft/sec^2 and a peak deceleration of 6 ft/sec^2. In a similar test documented in reference (d) the CH-53E decelerated from 170 knots to 45 knots in 40 seconds with an average deceleration of 5 ft/sec^2 and a peak deceleration of 10 ft/sec^2. Control excursions in both reference (c) and (d) were well within available limits. Since no complaints were noted about the deceleration capability of either the AH-1T or the CH-53E, it appears that a deceleration capability of 5 ft/sec^2 may be adequate unless specific mission requirements dictate a more severe maneuver. In any case some numerical deceleration specification should be added to the MIL-H-8501A quick stop requirement to make it more meaningful. In addition some limits on the order of ± 100 feet should be placed on the allowable altitude excursion during the quick stop maneuver.

2. Longitudinal Trim Stability

Paragraph 3.2.10: "The helicopter shall, at all forward speeds and at all trim and power conditions specified in Table I, except as noted below, possess positive, static longitudinal control force, and control position stability with respect to speed. This stability shall be apparent in that at constant throttle and collective pitch-control settings a rearward displacement of and pull force on the longitudinal-control stick shall be required to hold a decreased value of steady, forward speed, and a forward displacement and push force be required to hold an increased value of speed. In the *speed range between 15 and 50 knots forward, and* 10 to 30 knots rearward, the same characteristics are desired, but a moderate degree of instability may be permitted. However, the magnitude of the change in the unstable direction shall not exceed 0.5 inch for stick position or 1.0 pound of stick force."

TABLE I. Power and speed conditions

Initial trim and power condition	Speed range of interest
Hovering	0 to 30 knots.
Level flight at 35 knots	15 to 60 knots
Level flight at 80 percent V_{max}	60 percent V_{max} - V_{max}*
Level flight at V_{max}	80 percent V_{max} - V_{limit}*
Climb at best rate of climb	V_{max} R/C ± 15 knots.
Partial power descent at 300 to 500 fpm	15 to 60 knots.
Autorotation with trim as in "level flight at 80 percent V_{max}" above	60 percent V_{max} - V_{max} for autorotation.
Autorotation at speed for minimum rate of descent	15 knots - (trim speed + 20 knots)."

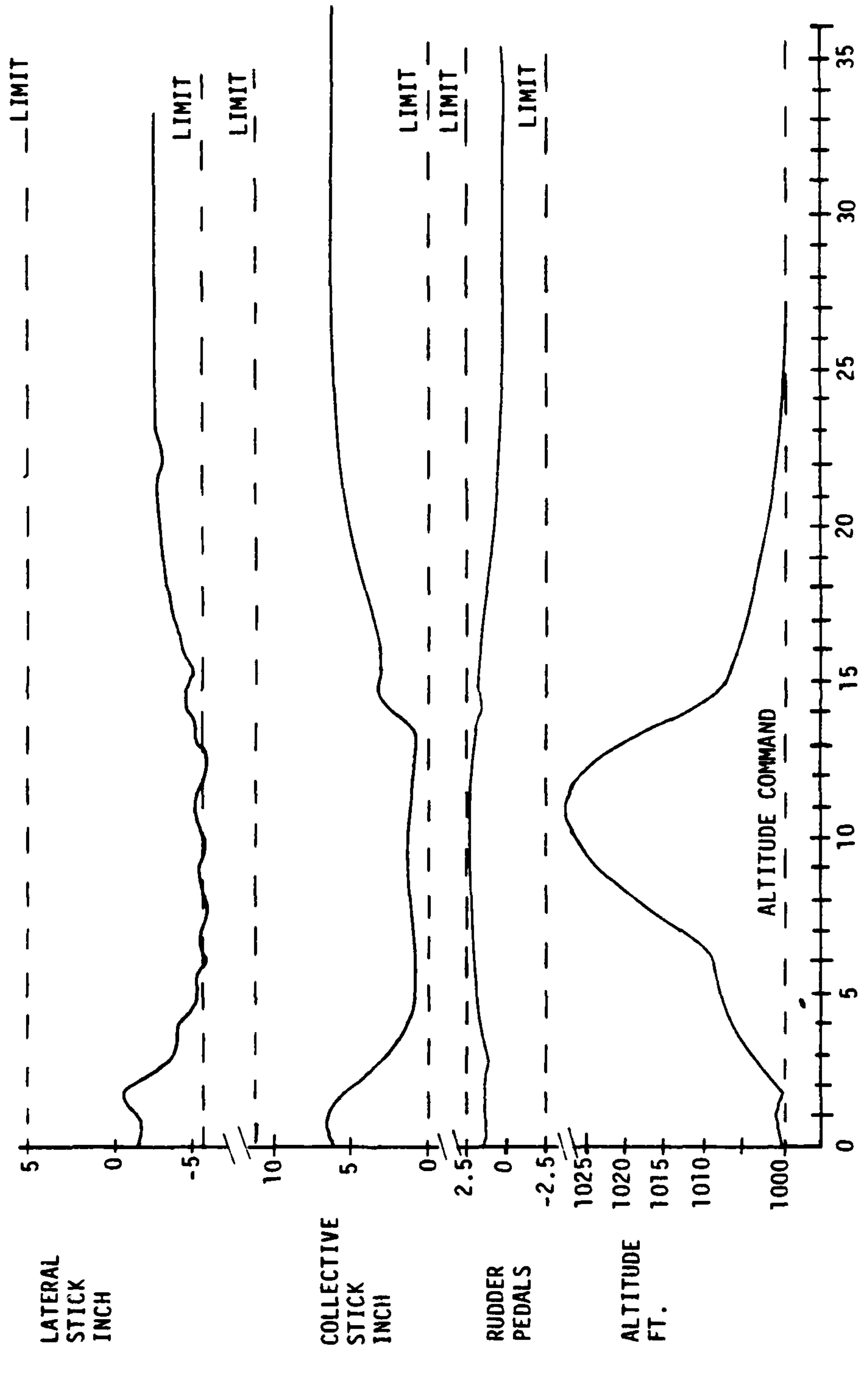

Figure 4. SH-2F Quick Stop Maneuver Using Velocity Ramp Command - Control Input

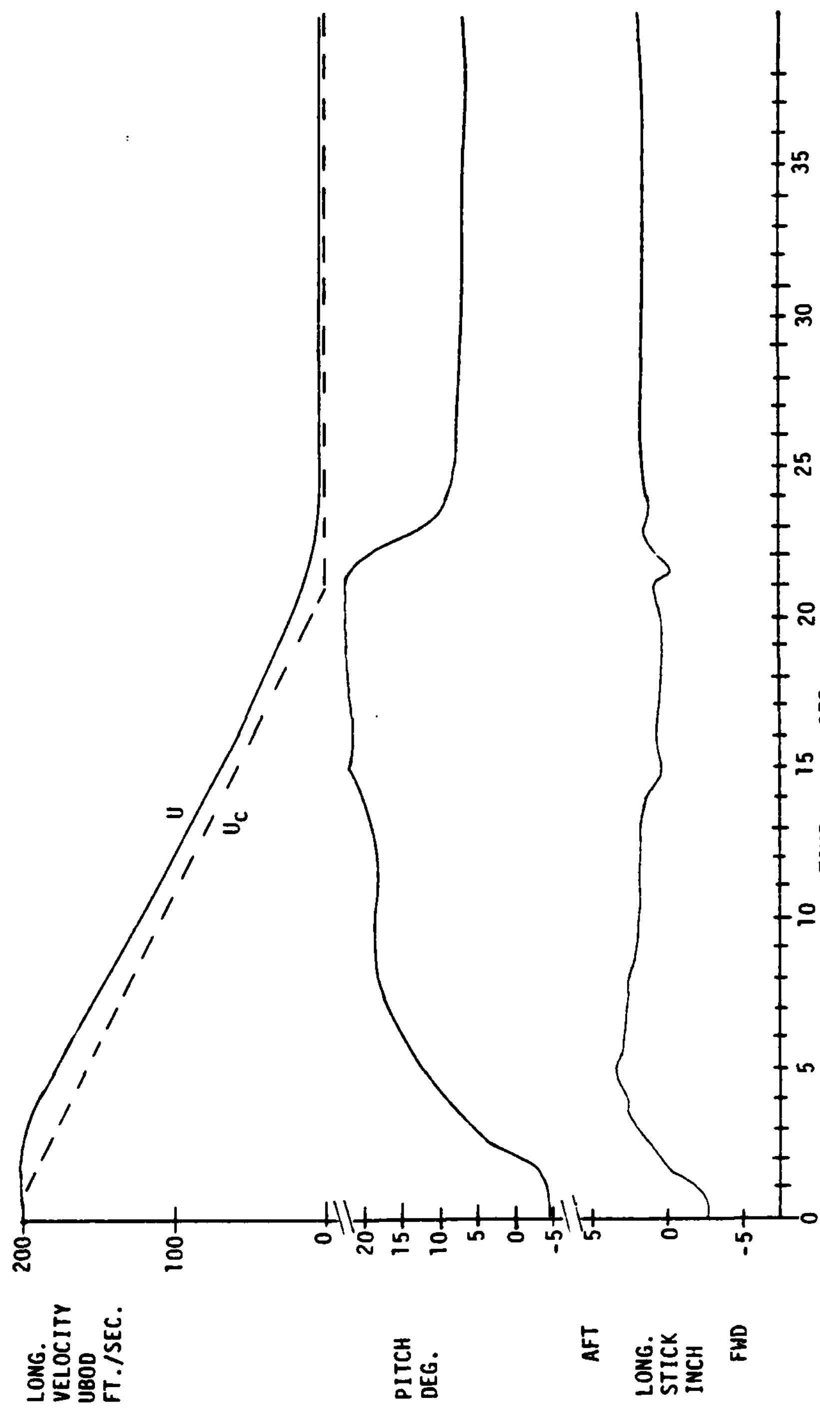

Figure 3. SH-2F Quick Stop Maneuver Using Velocity Ramp Command

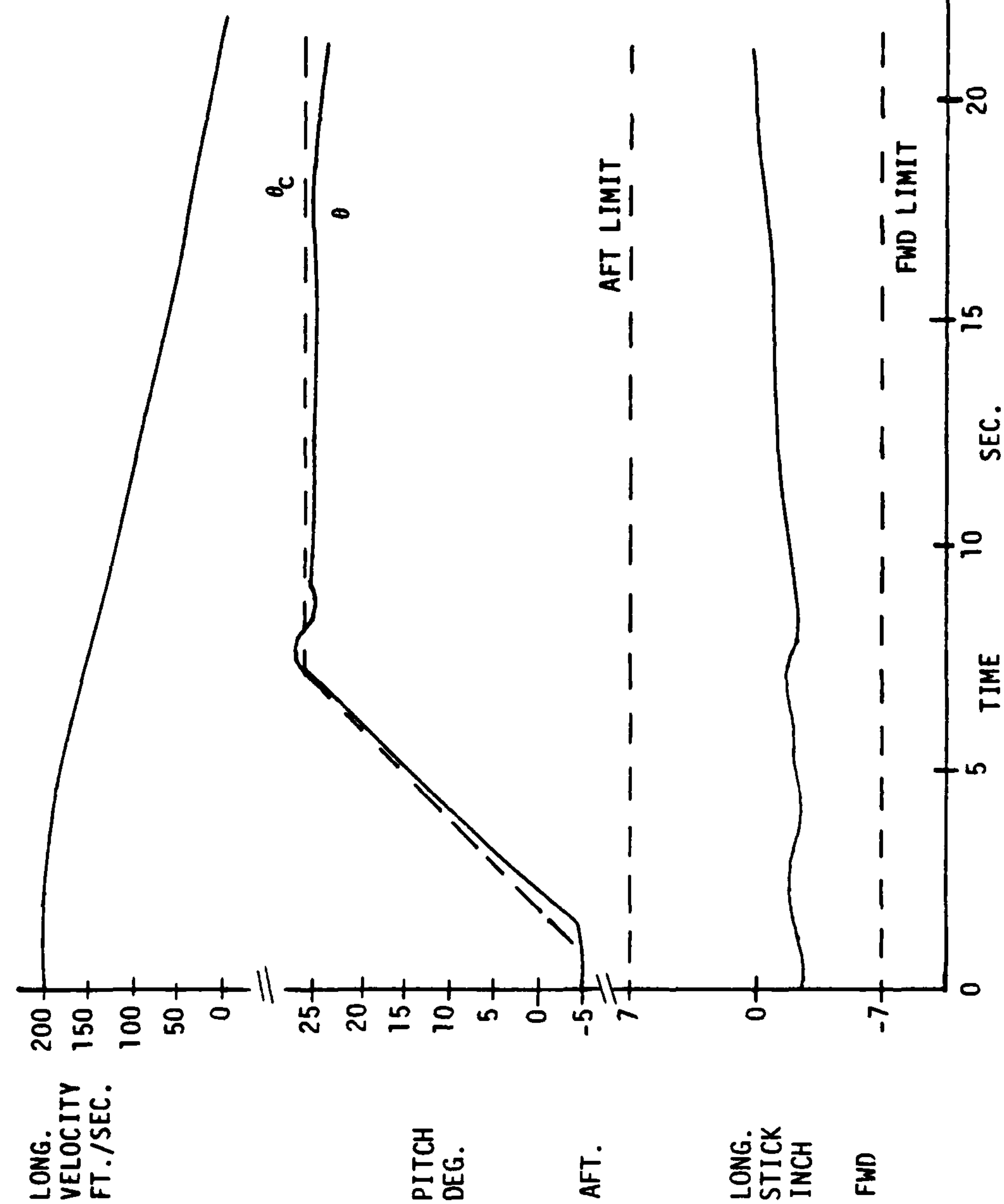

Figure 2. Simulated Quick Stop Maneuver - 30 degree pitch command with collective fixed

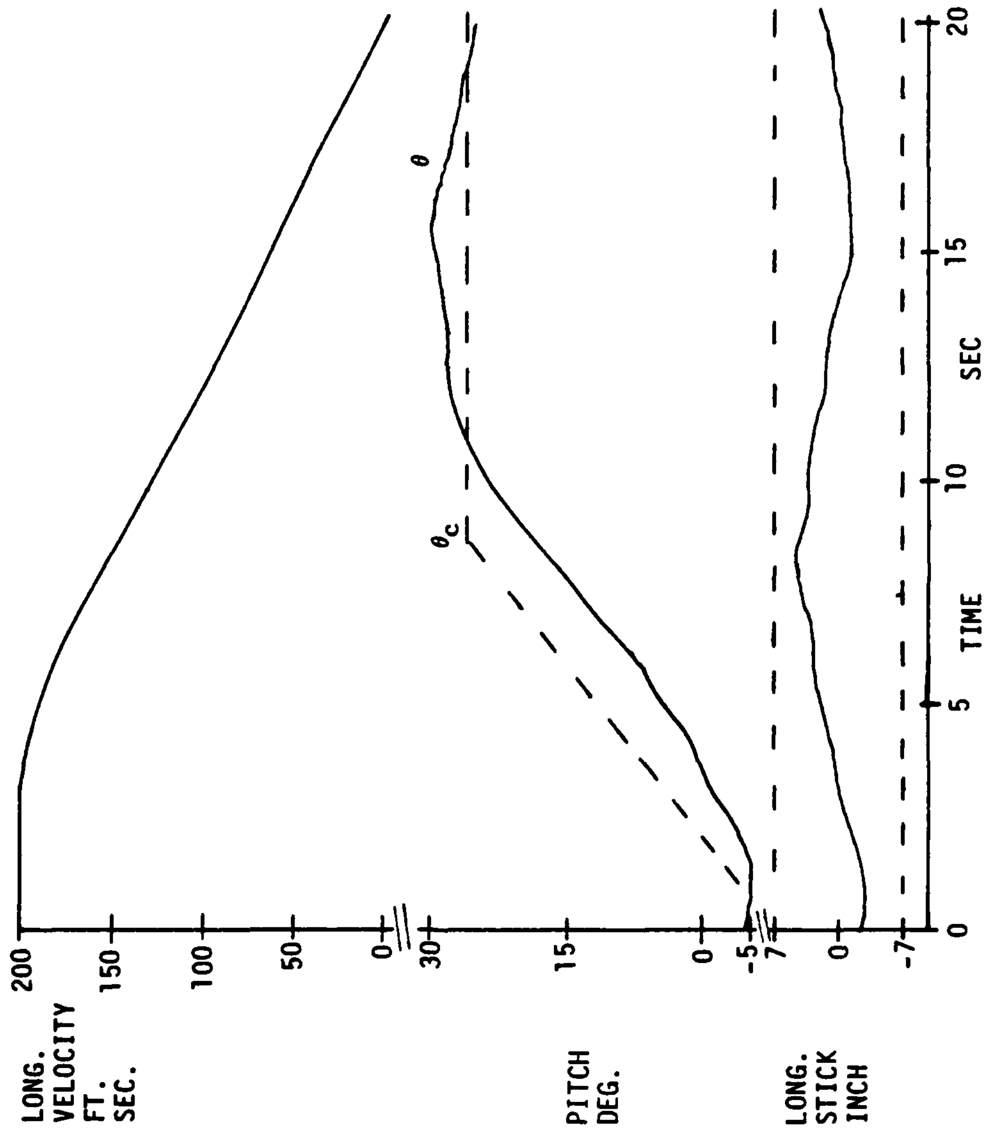

Figure 1. SH-2F Quick Stop from 120 knots - using 30 degree nose up ramp command

dangerous at low altitudes because of the possibility of striking the ground with the tail, as well as the possibility that the pilot will become disoriented due to loss of visibility at high pitch angles.

Two techniques were used to approximate pilot control inputs during the simulated quick stop maneuver. The attitude control system was driven by a pitch ramp with bank, heading, and altitude references held fixed. In an alternate approach the ground speed hold mode was engaged and the command speed was ramped from the initial value to zero. Because the trim longitudinal stick varies significantly with forward speed, the basic ASE System lacked sufficient control authority to accurately follow the commanded speed and attitude changes.

The pitch attitude error caused by changes in the longitudinal stick trim due to variation in airspeed was reduced by increasing the authority of the ASE to 100 percent available control travel and by adding a longitudinal stick trim follower. With the trim follower engaged, the incremental stick position commanded by the ASE due to pitch error is passed through a first order lag with a 5 second time constant and then added to the initial trim stick position. Response of the modified control system is illustrated in figure 1. The model response follows the pitch command ramp with a two second lag and then stabilizes near the maximum commanded attitude with a maximum overshoot of four degrees.

The 30 degree pitch ramp causes both the lateral stick and the collective stick to briefly exceed allowable control limits. Thus this test maneuver may exceed the practical limitations of the aircraft.

Large amounts of down collective were required by the model to avoid altitude gain at high speed. This collective control input in turn requires a substantial lateral stick command to correct for the induced roll coupling. Although no data are available for the SH-2F performing this maneuver, the large reduction in collective stick seems somewhat excessive.

The pitch ramp appears to be the simplest strategy for decelerating the aircraft. All control movements are smooth with very little high frequency oscillation. This is indicative of a moderate pilot workload. In actual flight the pilot would gradually lower the nose and establish a hover as the airspeed approached zero. However, the attitude control model lacked the capability of duplicating this more complex strategy.

Execution of the quick stop maneuver is complicated by the requirement to maintain constant altitude. In order to evaluate the effect of maintaining altitude, the pitch ramp command maneuver was repeated with the collective control held fixed. As shown in figure 2 the pitch attitude tracks the pitch command with very little error, and the longitudinal, lateral, and yaw control inputs are greatly reduced compared with those of figure 1. However, with collective fixed, the model shows an altitude gain of about 800 feet. Thus holding collective fixed is not a satisfactory strategy for the maneuver even though it greatly simplifies the task.

The ground speed hold mode was also tested as a technique for executing the quick stop maneuver. A constant deceleration command of 10 ft/sec^2 was used to drive the ground speed ASE mode. Altitude, heading, and lateral velocity hold were engaged and a longitudinal trim follower was implemented to minimize steady state error.

As figures 3 and 4 show, airspeed follows the commanded speed change fairly closely during the deceleration and that only a small steady state error remains at the end of the maneuver. Pitch attitude response is smooth with a maximum value of 22 degrees. Longitudinal stick travel remains well within maximum available travel and shows no rapid oscillations. Lateral stick reaches full left

Ground Speed Hold:

$$\Delta \text{ Lateral Stick} = -.121 \cdot \left[V_{Y_B} - V_{Y_{COM}}\right] + 1.819 \cdot [\phi_{ref} - \phi] - 1.05 \cdot P$$

Position Hold:

$$\Delta \text{ Lateral Stick} = .606 \cdot V_{Y_B} - .1212 \cdot Y_B + .606\,[\phi_{ref} - \phi] - .105 \cdot P$$

Yaw axis
Attitude Hold:

Airspeed < 50 knots
Feet off pedals

$$\Delta \text{ Pedal} = 0.175 \cdot [\Psi_c - \Psi] - .15 \cdot r$$

Airspeed > 50 knots

$$\Delta \text{ Pedal} = 0.35 \cdot [\Psi_c - \Psi] - .30 \cdot r$$

DISCUSSION OF THE REQUIREMENTS OF MIL-H-8501A AND A COMPARISON WITH THE SH-2F MODEL CHARACTERISTICS

1. Quick Stop Maneuver

Paragraph 3.2.5: "With the helicopter trimmed in steady, horizontal flight at maximum forward speed, it shall be possible readily and safely to bring the machine to a quick stop and hover."

No quantitative limits for deceleration are specified here. It implies, however, that the helicopter must be able to transition smoothly from maximum speed to any lower speed down to hover without encountering any control stops or requiring any unusual or difficult pilot control activity. In practice the deceleration is limited by the maximum nose up attitude that can be attained safely, and by the maximum variation in altitude that can be tolerated. Although paragraph 3.2.5 does not explicitly state it, the maneuver is normally performed at a nearly constant altitude.

Successful completion of the quick stop involves a compromise between increasing the pitch attitude to rotate the thrust vector aft and lowering the collective to offset the initial thrust build up produced by increasing the rotor disk angle of attack at high speeds.

The quick stop maneuver is useful in situations where it is desired to approach a point a maximum speed and then come to a full stop to deliver troops or supplies in minimum time without circling. The maneuver may also be useful in low altitude collision avoidance, but in most cases the pilot would turn or climb to avoid an obstacle in his path. It is an excellent test to prove that the helicopter is fully controllable in extreme maneuvers. However, the maneuver can be rather

It was necessary to modify the control laws in order to successfully complete all of the desired maneuvers. The control authority limits were increased to 100 percent in all axes. Even with full authority, the proportional controller could not maintain close tolerances on attitude and airspeed response to commanded changes. Most of the difficulty was caused by the large changes in trim longitudinal stick position with airspeed. A longitudinal trim follower and integral of airspeed error feedback were added to the pitch channel to reduce steady state errors.

A position hold controller was also developed to allow the execution of a turn over a spot maneuver. Position error was resolved into X and Y components along the body axes to determine the proper drive signals to the longitudinal and lateral control channels.

Pilot inputs during autorotation were simulated by commanding full down collective after a prescribed delay time. Collective inputs could also be combined with a pitch attitude command to increase or decrease airspeed. The control laws used for the study are summarized below:

Longitudinal Axis:
Pitch attitude hold:

$$\Delta \text{ Longitudinal Stick} = 0.47 \cdot [\theta_c - \theta] \quad - .058 \cdot Q$$

Ground speed hold:

$$\Delta \text{ Longitudinal Stick} = 0.47 \cdot [\theta_{ref} - \theta] \quad - .058 \cdot Q$$
$$+ 0.66 \cdot [U - U_c]$$
$$+ .066 \cdot \int [U - U_c] \; dt$$

Position hold:

$$\Delta \text{ Longitudinal Stick} = 0.47 \cdot [\theta_{ref} - \theta]$$
$$- .116 \cdot Q + 0.66 \cdot V_{X_B} + 0.13 \cdot X_B$$
$$+ .066 \cdot \int [U - U_c] \; dt$$

Longitudinal Stick Trim Follower:

$$\Delta \text{ Long. Trim Stick} = \left[\frac{0.2}{S+0.2}\right] \cdot \Delta \text{ Long. Stick}$$

$$\text{Total Longitudinal Stick} = \text{initial trim} + \Delta \text{ Long. trim} + \Delta \text{ Long. Stick}$$

Lateral Axis:
Attitude Hold:

$$\Delta \text{ Lateral Stick} = 1.53 \times [\phi_c - \phi] \quad -.529 \cdot P$$

INTRODUCTION

The SH-2F simulation model developed for the NAVTOLAND Program was evaluated by comparing the trim and dynamic response characteristics of the simulated helicopter against flight test data and against the requirements of MIL-H-8501A. A series of model trim calculations and simulated maneuvers were performed in order to evaluate the implications of the flying qualities specification requirements. Performance characteristics of the model were compared with all available flight test data for the SH-2F.

Reference (a), which is the current helicopter flying qualities specification, is over twenty years old and provides very limited guidance in many areas of helicopter dynamic characteristics. This study was undertaken to gain an understanding of the physical significance of the quantitative portions of the specification. The analysis also provided an opportunity to evaluate the performance of the simulation model in a variety of large amplitude maneuvers. Although the math model does not as yet exactly duplicate the real helicopter; it provides a flexible tool for examining a wide variety of maneuvers and trim conditions that could not be conveniently or safely examined in flight. It was particularly valuable for analyzing large perturbation maneuvers that could not be considered adequately through linear systems analysis, and which could be dangerous to demonstrate by flight test.

DESCRIPTION OF THE SH-2F SIMULATION MODEL

The SH-2F simulation model used in this analysis was developed in support of the NAVTOLAND VSTOL automatic landing system program. This model was developed in part from the description of the Navy SH-2F training device found in reference (b). A servo-controlled flap system located on the outboard trailing edge of each rotor blade is used to provide cyclic and collective control for the main rotor. The current simulation model has been extensively modified from the original version derived from reference (b). Fundamental errors in the equations were corrected and empirical adjustments were made in order to match available flight test data as closely as possible.

The SH-2F simulation is a fully coupled six degree of freedom nonlinear model which consists of a main rotor, tail rotor, fuselage, engine, and control system modules. Extensive use is made of table interpolation functions to determine the aerodynamic characteristics of the fuselage and rotors.

DESCRIPTION OF THE SIMULATION MODEL CONTROL LAWS

The Automatic Stabilization Equipment (ASE) System for the SH-2F includes a four axis limited authority controller with two primary operating modes. These include attitude hold and a ground referenced velocity hold mode. In addition, the pilot may select an altitude hold mode. Radar altitude hold is available for flight less than 1000 feet above ground level. Barometric altitude hold can be used for other flight modes.

In the attitude hold mode, the longitudinal axis is driven by pitch attitude and pitch rate feedback. Bank angle and roll rate error drive the lateral channel, and azimuth is stabilized by yaw and yaw rate feedback to the tail rotor.

In the ground speed mode doppler radar signals are used to drive the longitudinal and lateral control axes in addition to attitude and attitude rate feedback.

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LIST OF SYMBOLS (Continued)

Symbol	
V_O	Trim airspeed - knots
V_{XB}	Longitudinal ground speed component - ft/sec
V_{YB}	Lateral ground speed component - ft/sec
V_{YCOM}	Lateral commanded ground speed - ft/sec
W	Aircraft gross weight - lb
X_B	Aircraft longitudinal postion - ft
Y_B	Aircraft lateral position - ft
θ	Pitch Attitude - deg
θ_{ref}	Trim Pitch Attitude - deg
$\dddot{\theta}$	$\frac{d\,\Theta^3}{d\,t^3}$ - Time rate of change of pitch acceleration - deg/sec^3
ϕ	Bank angle - deg
ϕ_C	Commanded bank angle - deg
ϕ_{ref}	Trim bank angle - deg
ψ	Aircraft heading - deg
$\ddot{\psi}$	Yaw acceleration - rad/sec^2
ψ_C	Commanded heading - deg
$\int$	Integral function

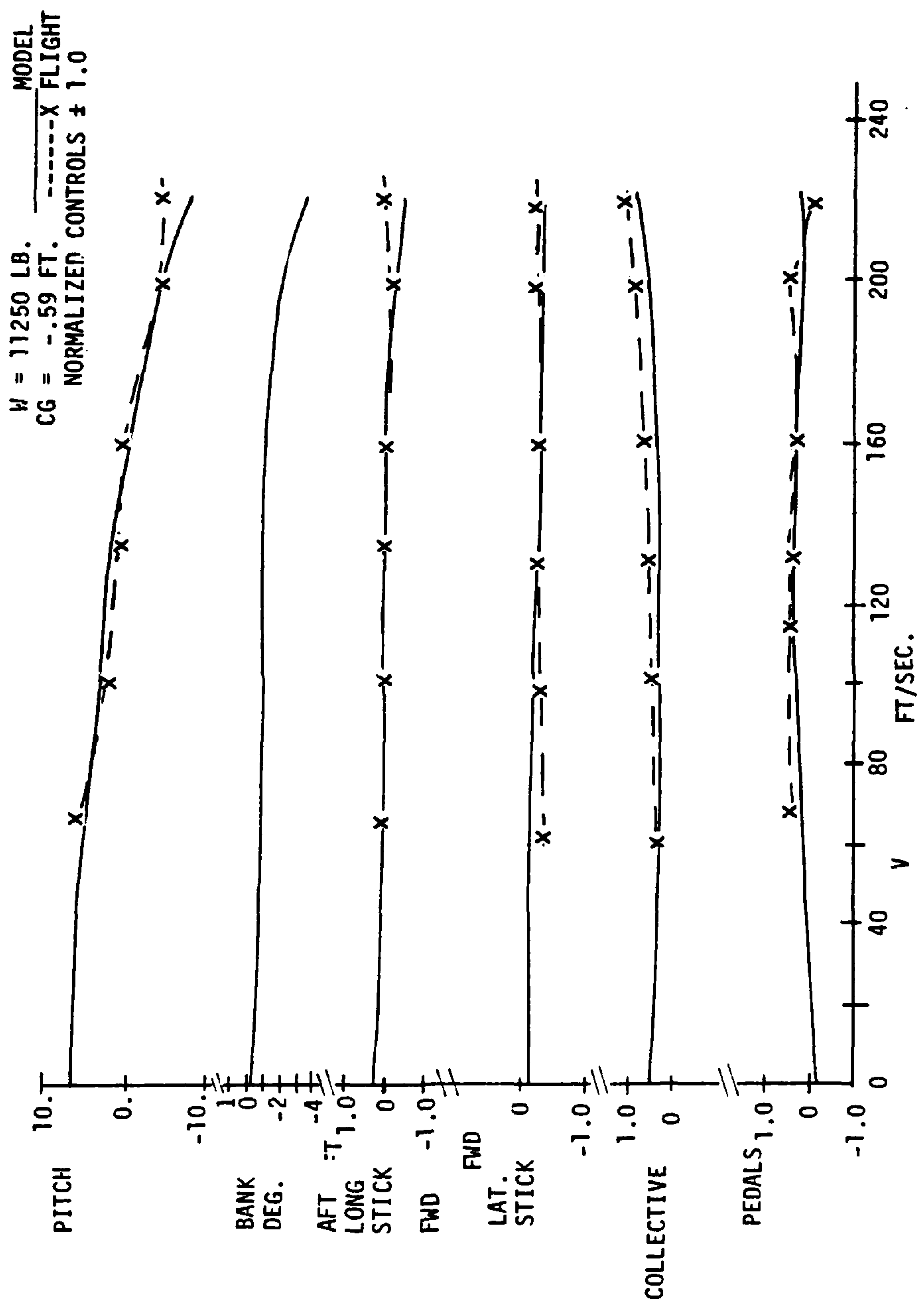

Figure 6. SH-2F High Speed Trim Characteristics vs. Flight Test

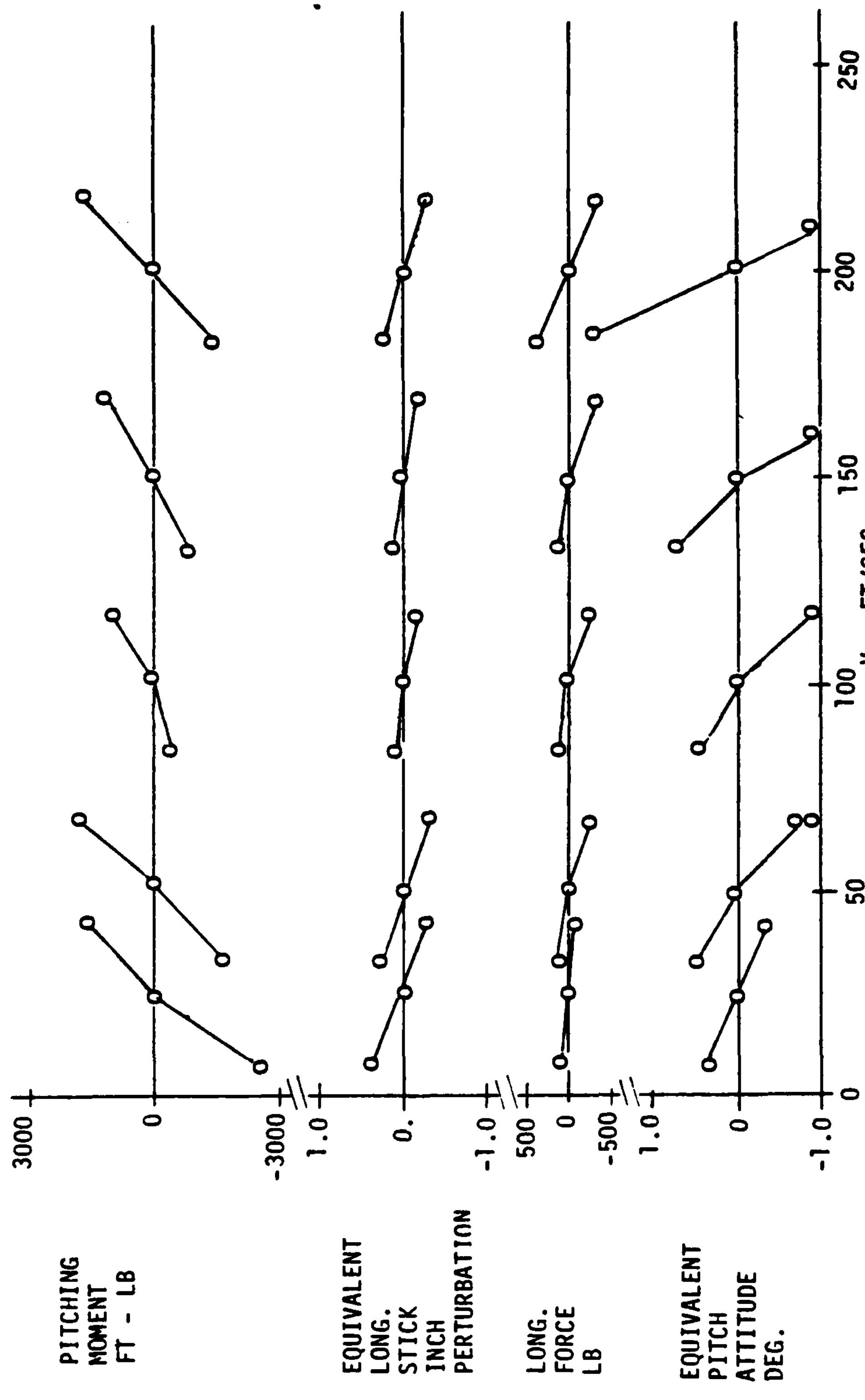

Figure 7. SH-2 Model Pitching Moment and Horizontal Force vs. Speed Perturbation with Controls Fixed

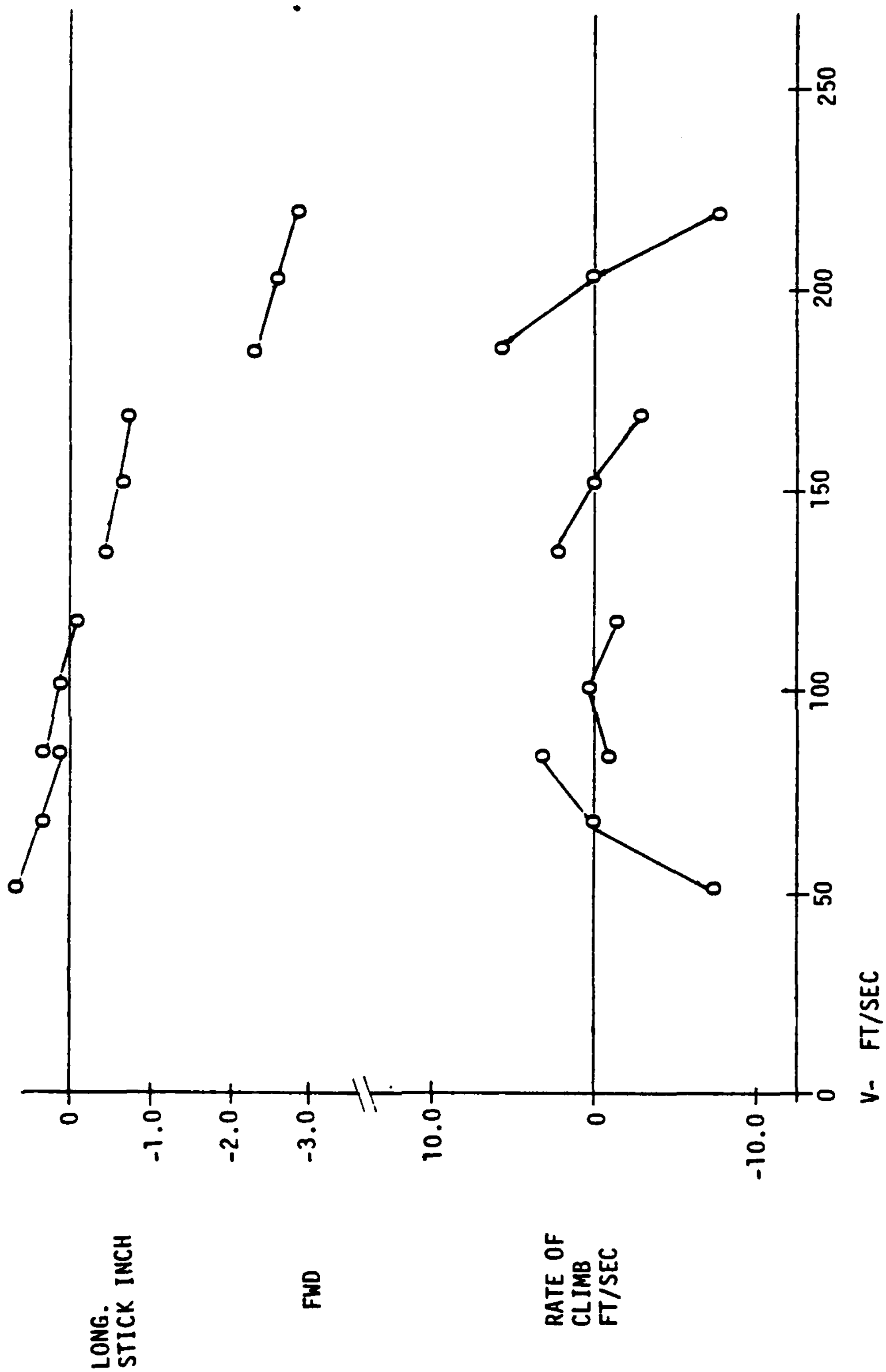

Figure 8. SH-2F Longitudinal Speed Stability and Sink Rate vs. Forward Speed with Fixed Collective Control - Model Response

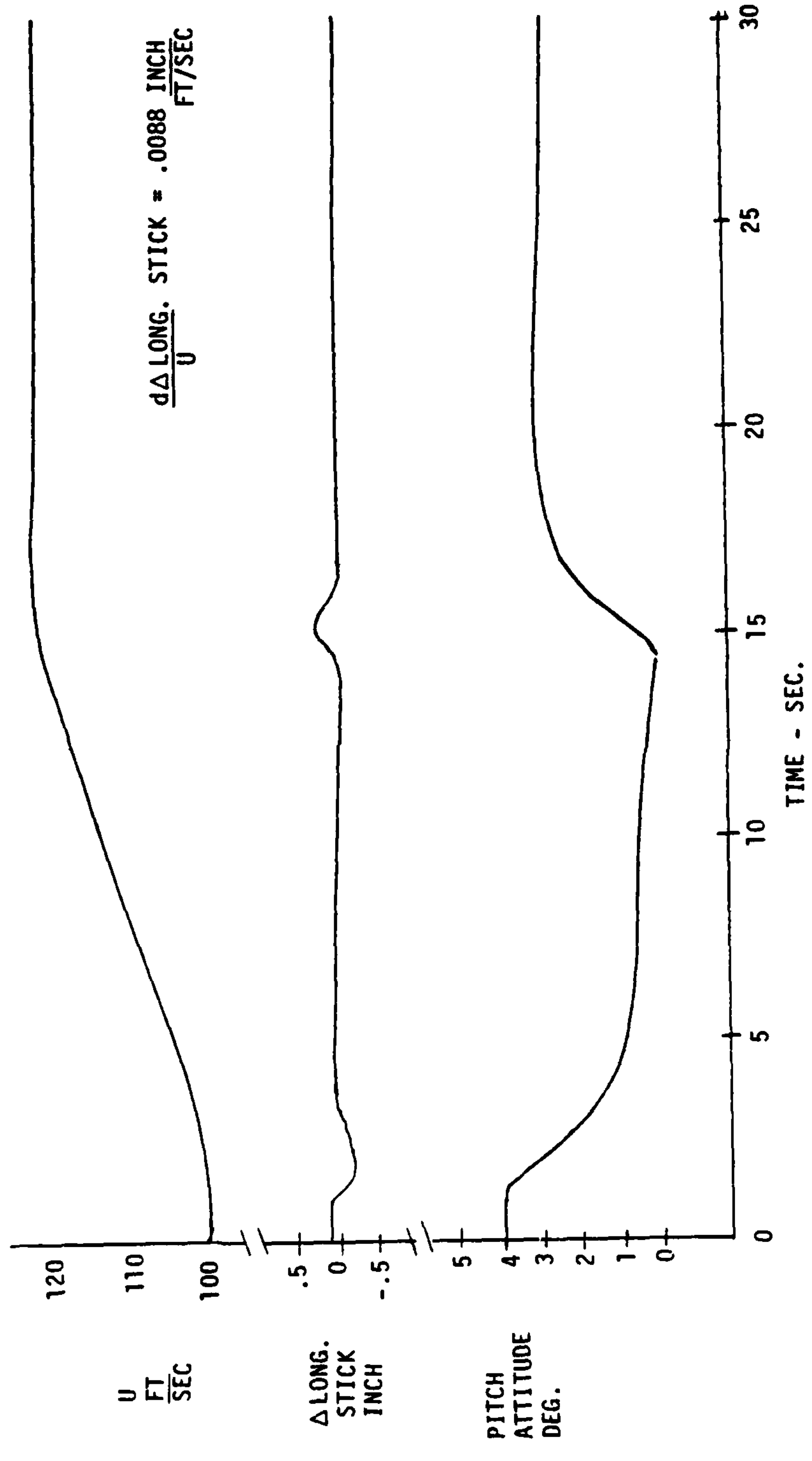

Figure 9. SH-2F Model Speed Stability Dynamic Response, V_o = 60 knots - Collective Fixed

ASE longitudinal control input in equivalent inches of stick and then adding the trim stick position. The time history calculation was continued until nearly constant velocity was achieved. The speed gradient was then calculated as the final minus initial equivalent longitudinal stick position divided by change in airspeed. Gradual speed change command ramps were used to avoid large control inputs. Figure 9 shows a typical result obtained from the model.

A 20 ft/sec velocity increase was commanded through the ground speed hold system using a ramp corresponding to 1.5 ft/sec^2 acceleration. The simulation shows a very slightly stable net change in longitudinal stick position which satisfies the specification. However, it seems unlikely that such a small gradient could provide very strong cues to the pilot. It is probably that the short term pitch response and stick dynamics may be more important than the steady speed gradient. If the dynamics are well-damped, the pilot could probably tolerate a significantly unstable steady stick gradient. Additional analysis is needed to determine the relationship between dynamic stability and static stick position gradient. It may be possible to establish an overall acceptance criteria that combines both aspects of the control response characteristics.

3. Longitudinal Trim Variation with Collective Control Position and Rate of Climb

Paragraph 3.2.10.2: "The helicopter shall not exhibit excessive longitudinal trim changes with variations of rate of climb or descent at constant airspeed. Specifically, when starting from trim, at any combination of power and airspeed within the flight envelope, it shall be possible to maintain longitudinal trim with a longitudinal control displacement of no more than three inches from the initial trim position as the engine power or collective pitch, or both, are varied throughout the available range. Generally, the airspeeds needing the most specific investigation of the above characteristics include V_{max} and the speeds between zero and one-half the speed for minimum power."

The model was trimmed in level flight at a range of speeds. Collective stick and thrust were varied over the largest range for which the pitching moment could be balanced by adjusting the longitudinal stick position. This served to evaluate the controllability of the helicopter following the sudden application of collective control. The model was also trimmed at vertical velocities covering the range from autorotation to maximum power climb in order to determine the steady state response to collective control.

The model was trimmed in level flight at speeds of 48, 77, and 107 knots. Thrust was varied at zero sink rate. These cases were compared to flight data taken at the same speeds. As shown in figure 10, the aircraft satisfies the requirement of having less than three inches of longitudinal stick movement about trim as the collective is varied from full up to full down. In contrast, the model shows stronger coupling between longitudinal and collective stick. However, the model control positions were computed in accelerated flight with zero rate of climb. This would correspond to a sudden application of collective in flight, and thus may be an extreme test. In actual flight, the pilot would gradually apply collective and allow the climb rate to build up rather than using a step input.

The model was next trimmed over a range of climb rates from autorotation to maximum power climb. The model showed moderate change in longitudinal stick position with climb rate at a trim speed of 48 knots. At a speed of 77 knots, the model just meets the three inch stick trim change specification. Stick trim change slightly exceeds allowable limits at 107 knots. In general, the model showed less variation in longitudinal trim with climb rate than with collective variation at zero climb rate. These calculated values agree well with the flight data. Figure 11 summarizes the trim longitudinal and collective stick of the model as a function of climb rate and forward speed.

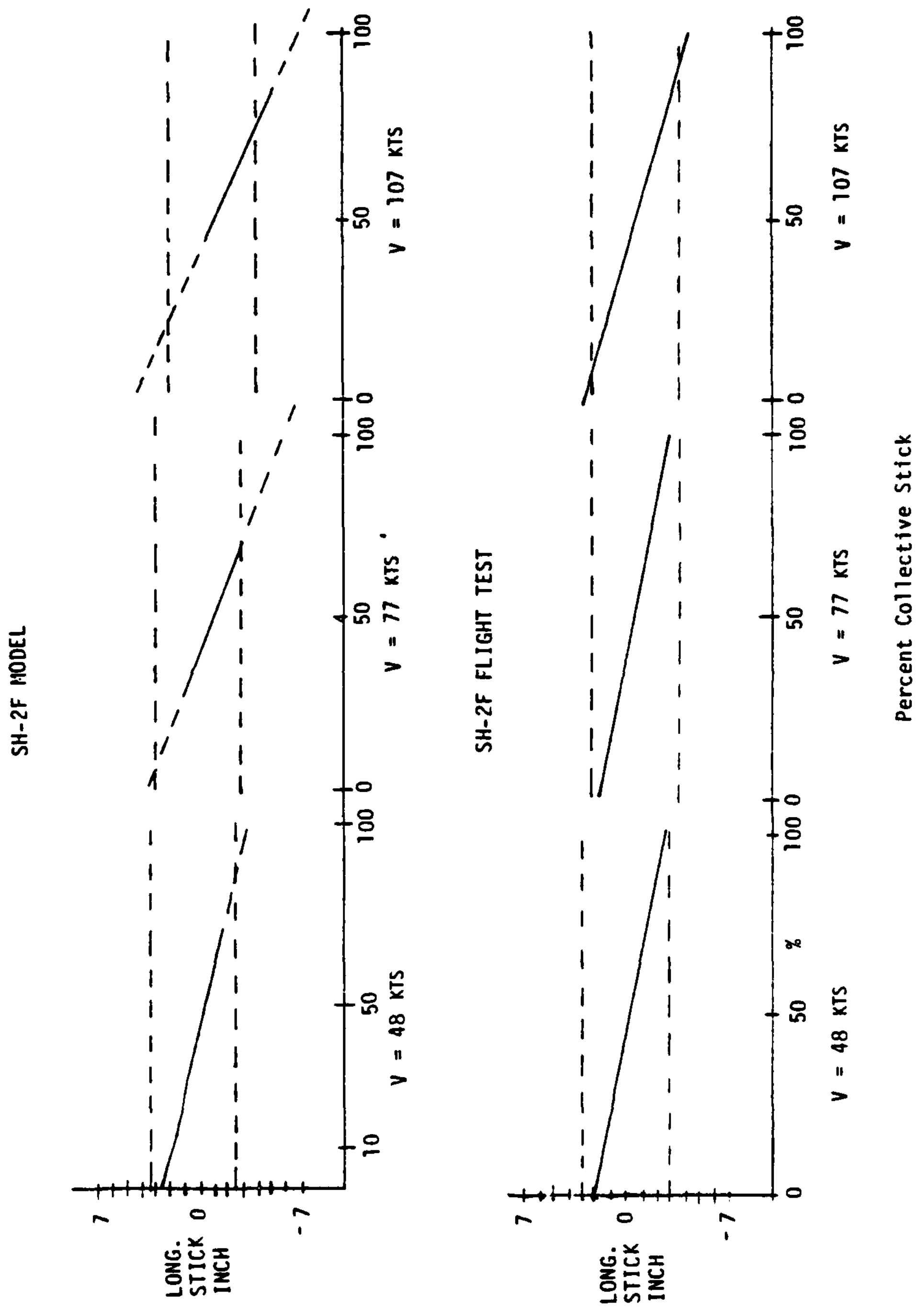

Figure 10. SH-2F Longitudinal Stick vs. Collective Stick from Model and Flight Test

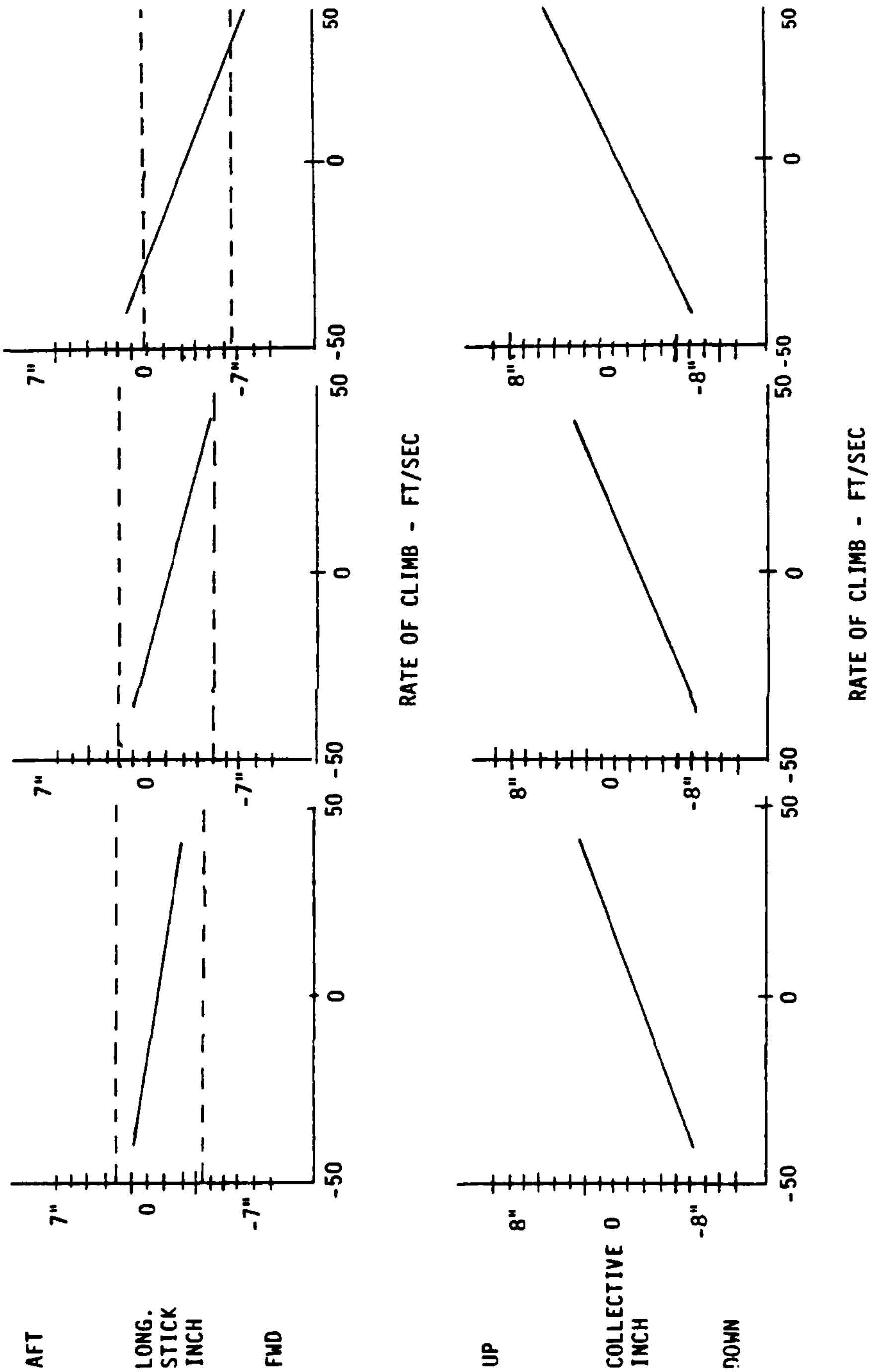

Figure 11. SH-2F Model Trim vs. Rate of Climb

The trim analysis indicated that the model had inaccurate autorotation characteristics at low speed. As the sink rate was increased for the 48 knot case, the power required decreased to a minimum value greater than zero and then increased for greater values of sink rate without passing through zero. In reality, the helicopter should be able to establish a stable autorotative descent starting from any forward speed. At higher speeds, the model autorotative characteristics were more realistic. The model achieved a steady autorotation at a sink rate of 39 ft/sec at a forward speed of 77 knots. This agrees well with the flight data found in reference (f).

4. Longitudinal Dynamic Stability Requirements

Paragraph 3.2.11: "The helicopter shall exhibit satisfactory dynamic stability characteristics following longitudinal disturbances in forward flight. Specifically, the stability characteristics shall be unacceptable if the following are not met for a single disturbance in smooth air:

(a) Any oscillation having a period of less than five seconds shall damp to one-half amplitude in not more than two cycles, and there shall be no tendency for undamped small amplitude oscillations to persist.

(b) Any oscillation having a period greater than five seconds but less than ten seconds shall be at least lightly damped.

(c) Any oscillation having a period greater than ten seconds but less than 20 seconds shall not achieve double amplitude in less than ten seconds."

The specification does not give specific details for the type of disturbances that should be introduced. Also, the speed range is not precisely defined. Forward flight implies any condition other than hovering or rearward flight. Table I describes hovering as any speed between 0 and 30 knots. Therefore, it may be reasonably inferred that forward flight implies all speeds of 30 knots and above. Longitudinal disturbances imply either longitudinal or collective control inputs. The dynamic characteristics of the model were examined by introducing longitudinal and collective stick doublet inputs for speeds of 30, 60, 90, and 120 knots. Control input amplitudes corresponding to ± five percent of full throw were selected to excite the helicopter response. Figures 12-16 illustrate the dynamic response characteristics of the helicopter model. At low speed, there is very little pitch coupling introduced by the collective control. The short-term response to collective input is a well damped oscillation in attitude and vertical velocity. At 30 knots, the longitudinal doublet excites a divergent oscillation with a period of about 18 seconds. An apparently aperiodic divergence develops after about 30 seconds. This divergence appears to result from the kinematic coupling between yaw rate and Euler pitch rate. That is:

$$\Delta\dot{\Phi} = r \cdot \tan(\theta), \Delta\dot{\theta} = -r \cdot \sin(\Phi).$$

A divergent oscillation with a period of approximately 20 seconds is evident in figure 13. However, the maximum amplitude achieved in the first 40 seconds is substantially smaller than in figure 12. It is apparent that all axes are excited by the collective input so that it is difficult to define a purely longitudinal oscillation. Higher frequency modes were not excited by the control doublet. Therefore, it may be assumed that any such modes are well damped. These modes might be identified in simulation studies by introducing a sinusoidal input of appropriate frequency. Identifying mode shapes from flight data is difficult because higher frequency modes tend to blend together and long term characteristics may be confused by turbulence or other disturbances.

As speed increases, the short term dynamics become more damped although an aperiodic divergence persists. The maximum amplitude achieved in 40 seconds decreases as the airspeed

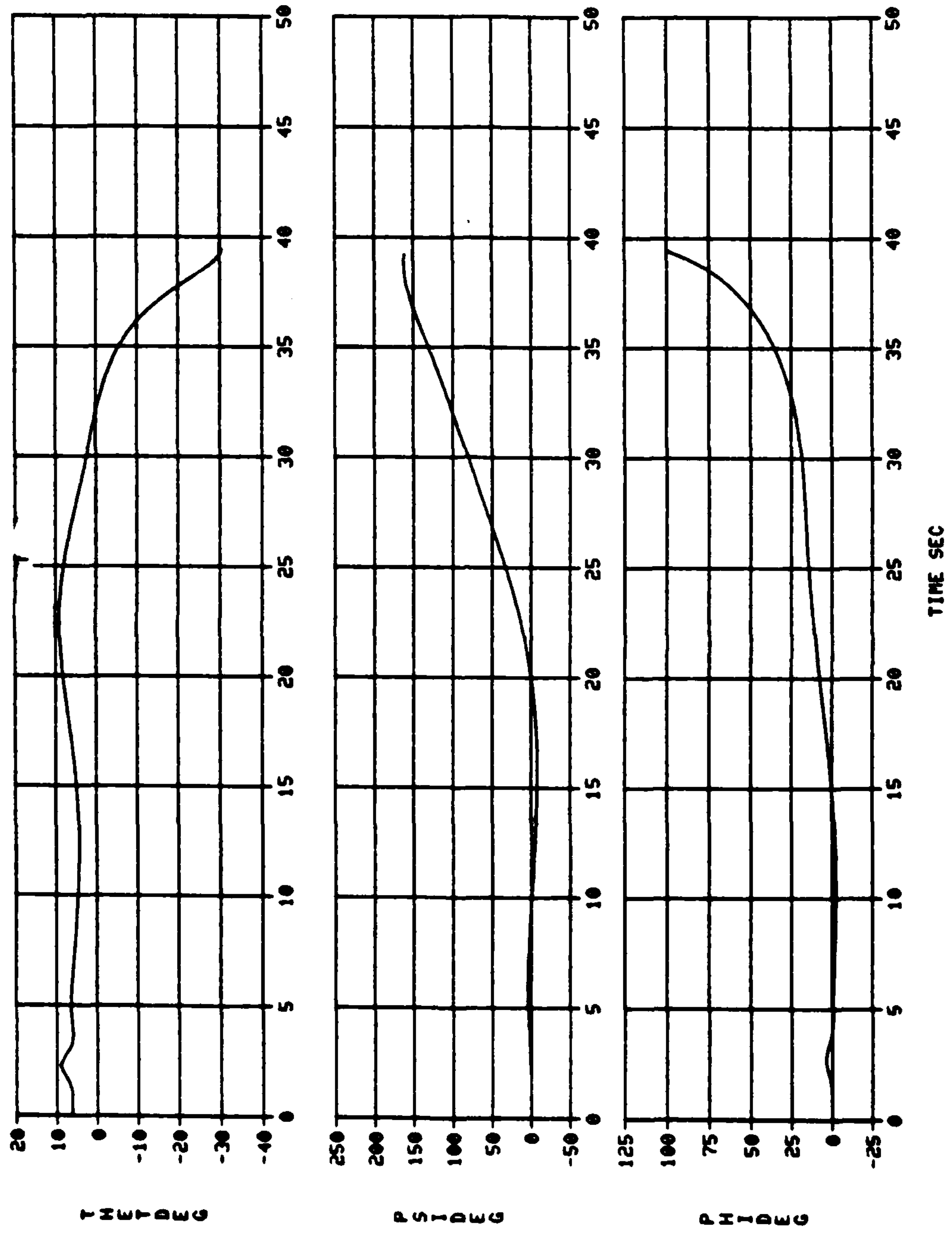

Figure 12. SH-2F Model Response to Longitudinal Control Doublet, $V_o = 30$ knots, ASE off

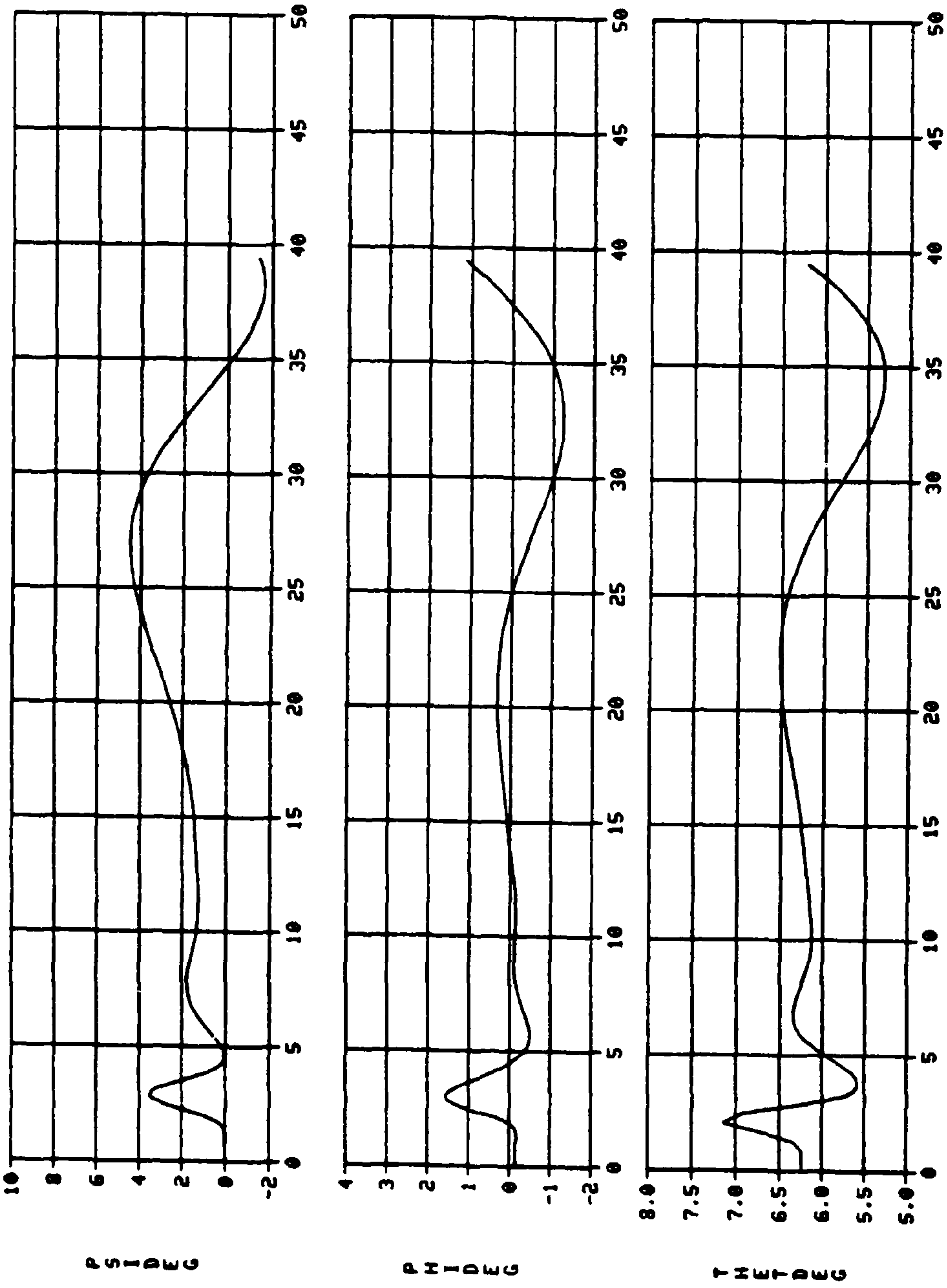

Figure 13. SH-2F Model Response to Collective Control Doublet, V_o = 30 knots, ASE off

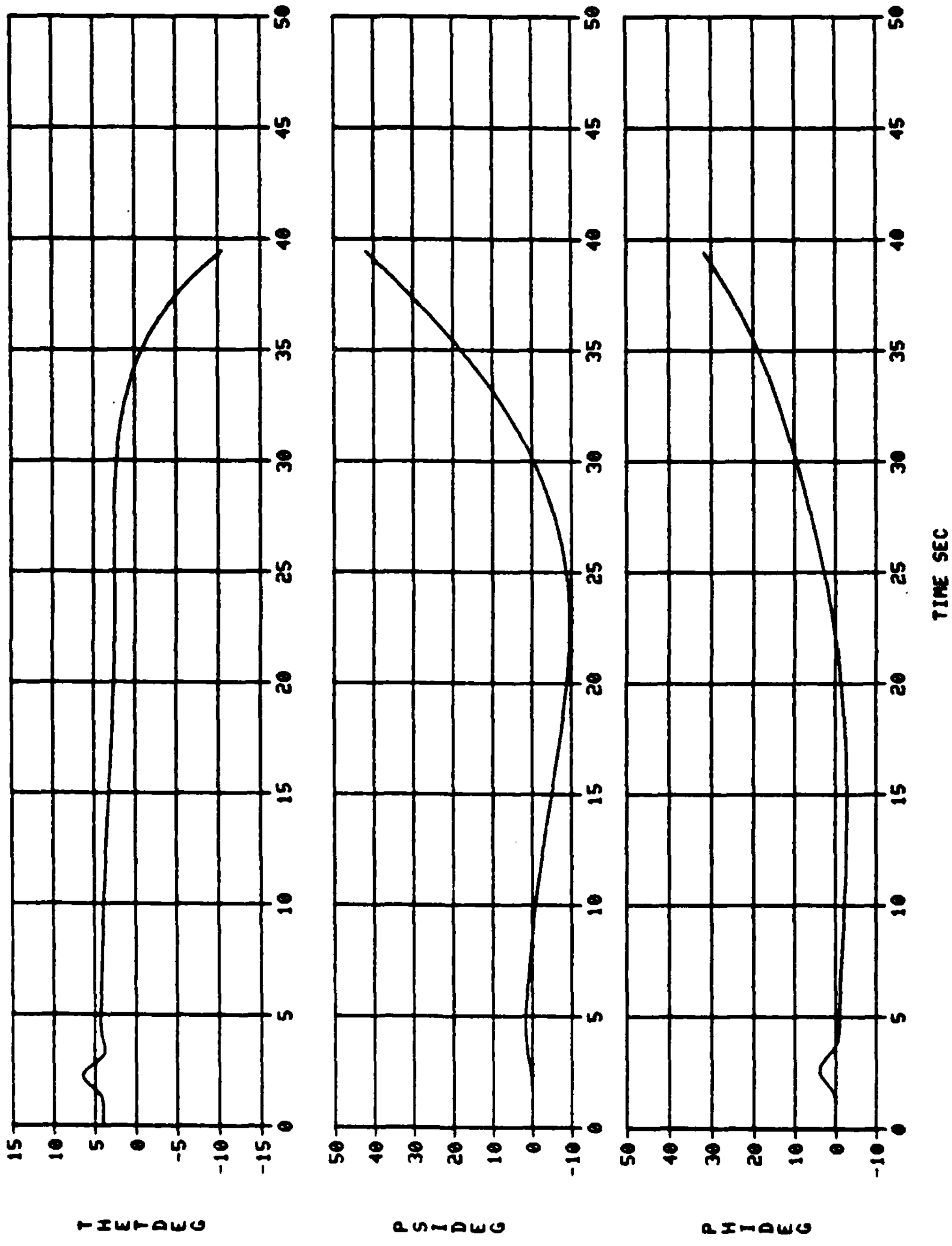

Figure 14. SH-2F Model Response to Longitudinal Control Doublet, ASE off, V = 60 knots

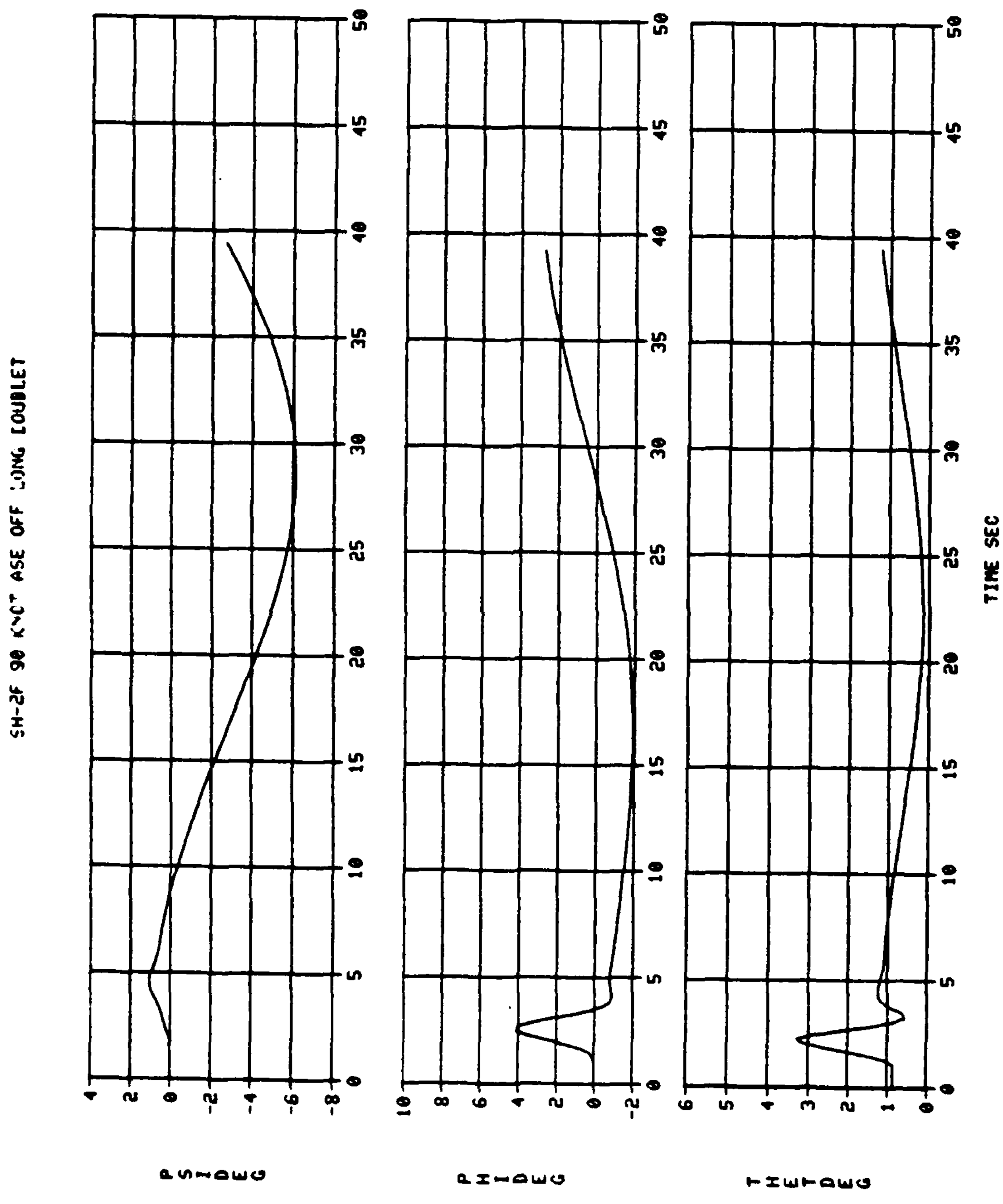

Figure 15. SH-2F Model Response to Longitudinal Control Doublet, ASE off, V = 90 knots

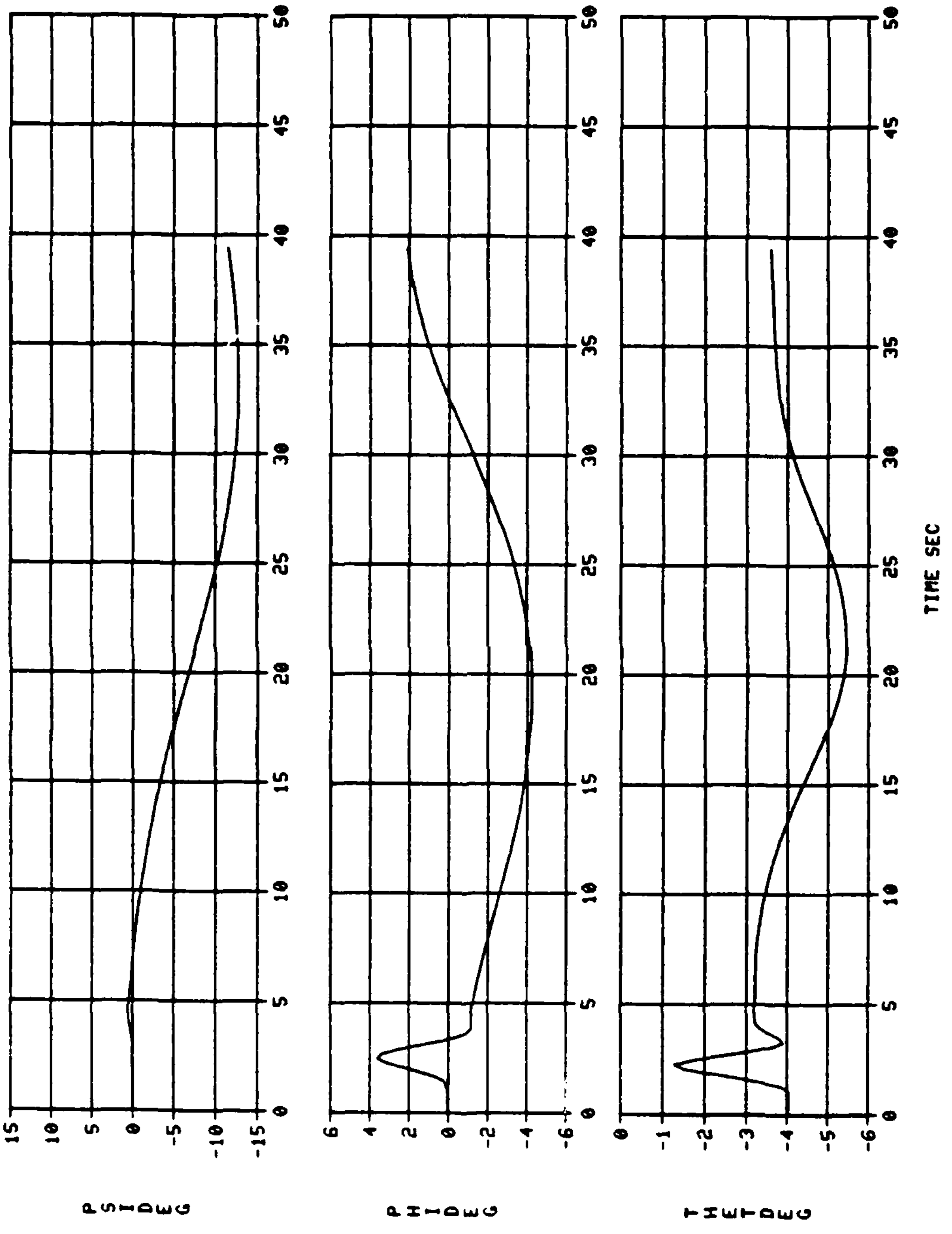

Figure 16. SH-2F Model Response to Longitudinal Doublet, V = 120 knots, ASE off

increases to 90 knots. Of the three attitude responses, heading shows the greatest reduction in amplitude as air speed is increased. Roll due to pitch coupling becomes a dominant factor in the dynamics for speeds above 90 knots. The simulation model appears to satisfy the intent of the specification for all speeds above 30 knots. However, it is very difficult to identify individual mode shapes needed to exactly determine compliance.

The specification does not address aperiodic divergence. Requirements for long period oscillations should be generalized to prohibit amplitude doubling in less than a prescribed period for aperiodic modes. The requirement for longitudinal disturbances should probably be generalized to include any disturbance since all degrees of freedom can be strongly coupled.

5. Concave Downward Requirement

Paragraph 3.2.11.1: "The following is intended to insure acceptable maneuver stability characteristics. The normal acceleration stipulations are intended to cover all speeds above that for minimum power required; the angular velocity stipulations shall apply at all forward speeds, including hovering.

(a) After the longitudinal control stick is suddenly displaced rearward from trim a sufficient distance to generate a 0.2 radian/sec. pitching rate within two seconds, or a sufficient distance to develop a normal acceleration of 1.5 g within three seconds, or one inch, whichever is less, and then held fixed, the time-history of normal acceleration shall become concave downward within two seconds following the start of the maneuver, and remain concave downward until the attainment of maximum acceleration. Preferably, the time-history of normal acceleration shall be concave downward throughout the period between the start of the maneuver and the attainment of maximum acceleration. Figure 17 is illustrative of the normal acceleration response considered acceptable.

(b) During this maneuver, the time-history of angular velocity shall become concave downward within 2.0 seconds following the start of the maneuver, and remain concave downward until the attainment of maximum angular velocity; with the exception that for this purpose, a faired curve may be drawn through any oscillations in angular velocity not in themselves objectionable to the pilot. Preferably, the time-history of angular velocity should be distinctly concave downward through the period between 0.2 second after the start of the maneuver and the attainment of maximum angular velocity. Figure 17 is illustrative of the angular velocity response considered acceptable."

The concave downward requirement is intended to protect against overcontrol of the aircraft. It guarantees a maximum upper limit to the short-term pitch rate and normal acceleration to a given pilot input. It does not, however, guarantee long-term stability of the aircraft.

The concave downward requirement was examined in hover and at forward speeds of 60 and 120 knots. Figure 18 shows that for hover a one-inch stick input determines the minimum condition specified by MIL-H-8501A. Maximum pitch rate was 5.5 degrees/second and normal acceleration was virtually unchanged initially. Determination of the concave downward requirement for pitch rate requires the calculation of the third derivative of pitch attitude $\dddot{\theta}$. In working with the simulation model, this was accomplished by taking differences in succeeding values of pitch rate and then dividing by the time increment. This was repeated with the values of pitch acceleration to yield the third derivative. The point of inflection is defined as the time where the third derivative of pitch attitude, $\dddot{\theta}$, changes from positive to negative. For the SH-2F model, the pitch response becomes concave downward about 0.4 seconds after the longitudinal stick input as seen in figure 18.

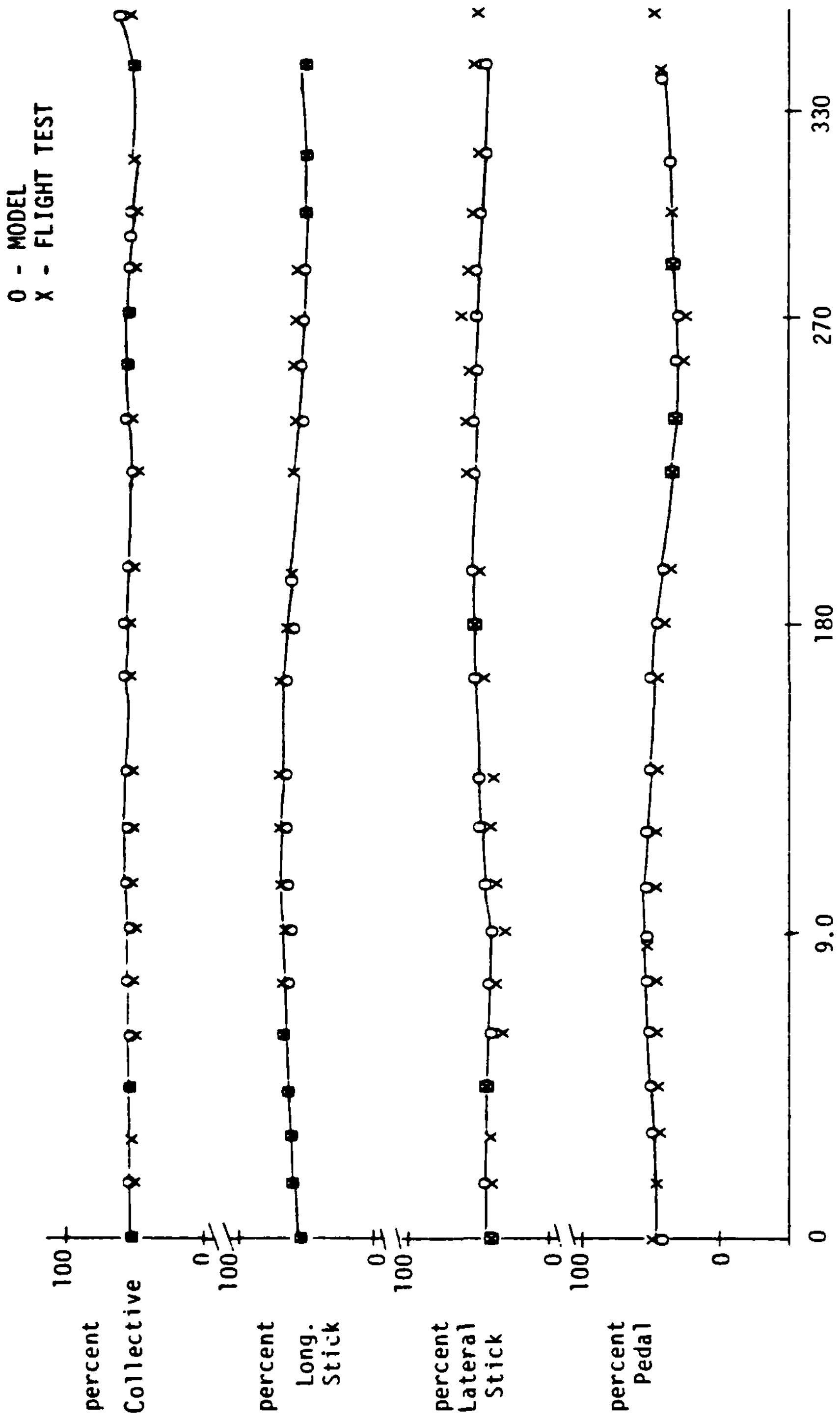

Figure 24. Critical Azimuth Sweep, SH-2F Model vs. Flight Test, V = 15 knots

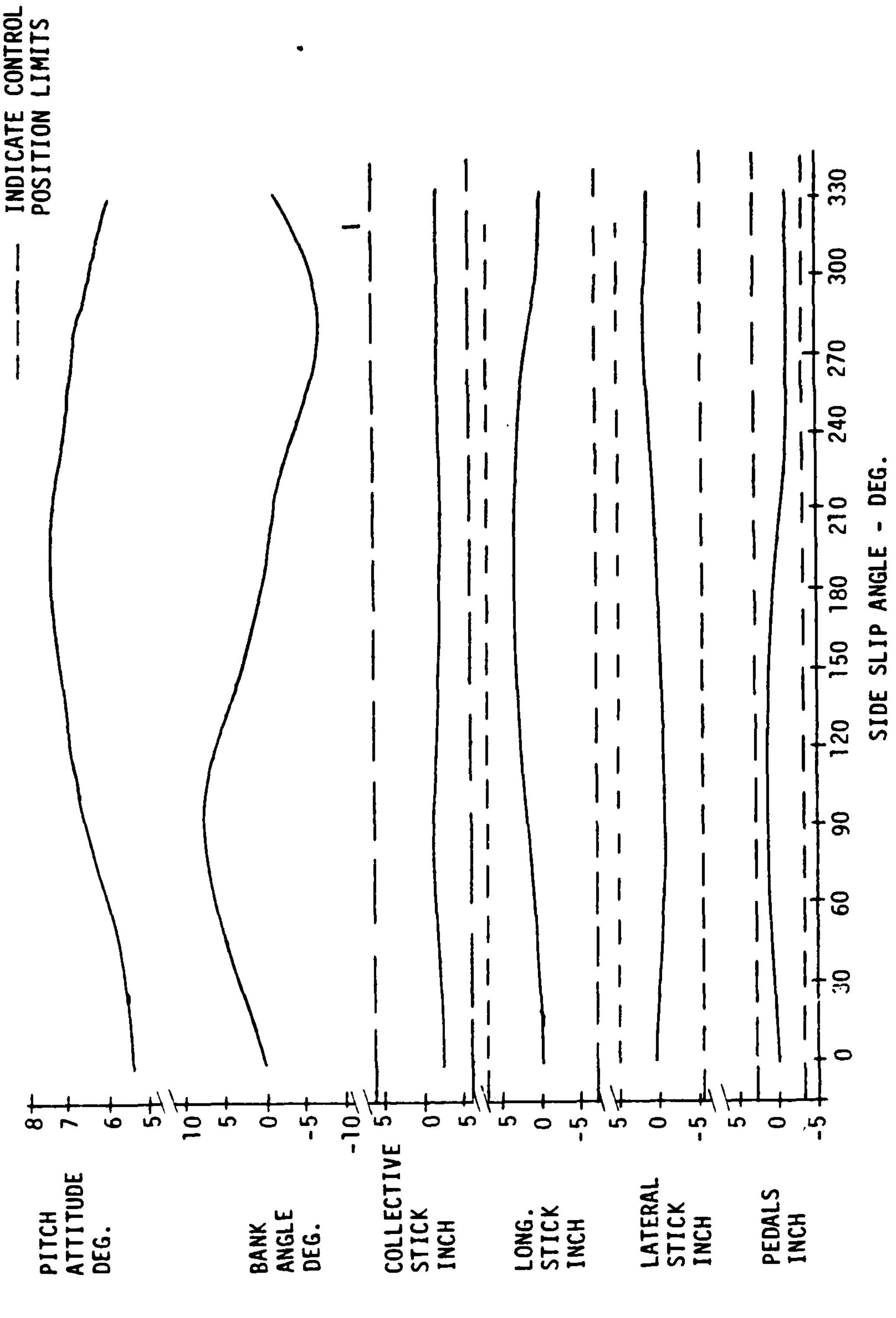

Figure 23. SH-2F Model Trim Characteristics vs. Sideslip Angle, V = 35 knots

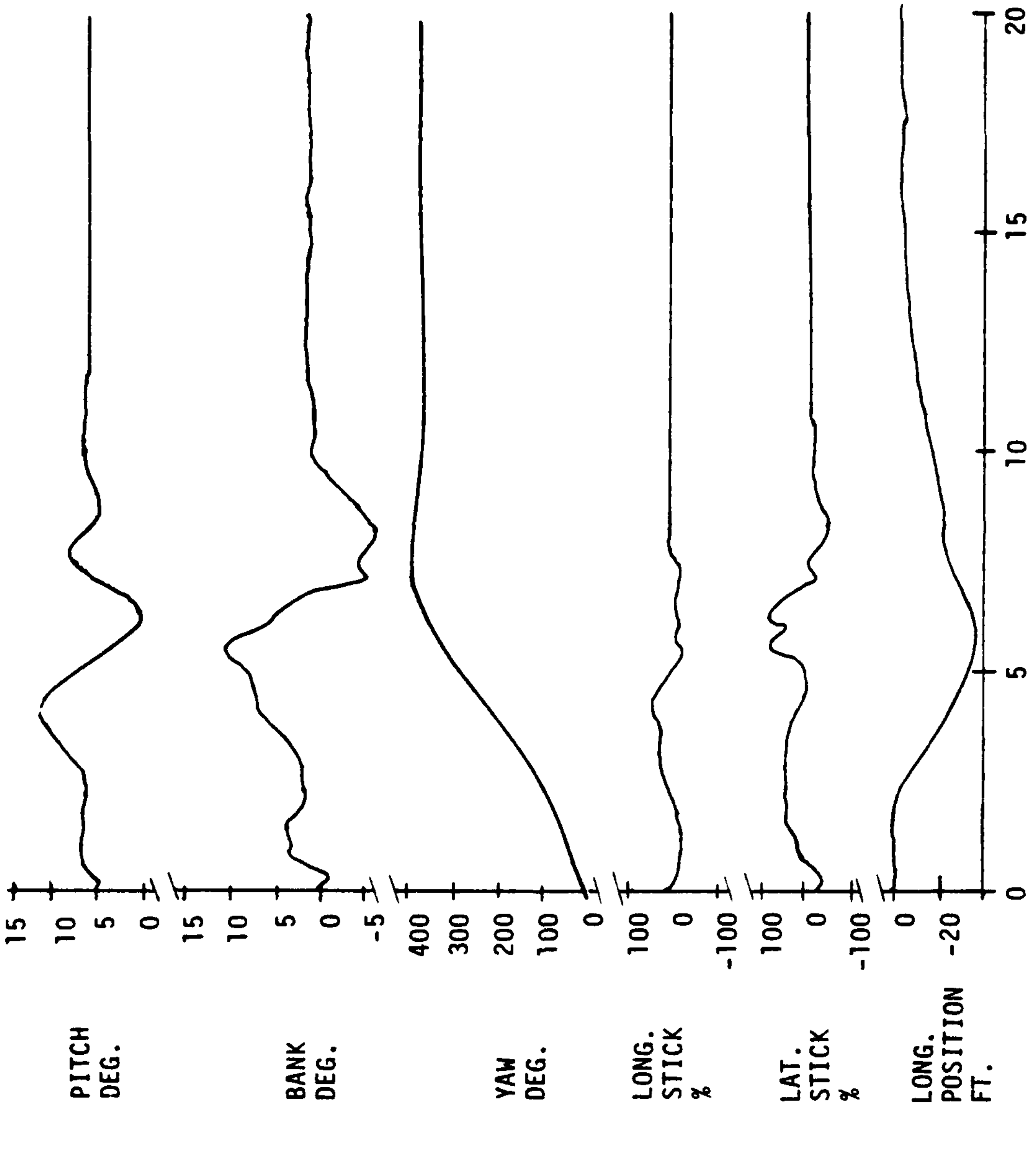

Figure 22. Model Response to a 360 Degree Turn Over a Spot with 35 knot wind

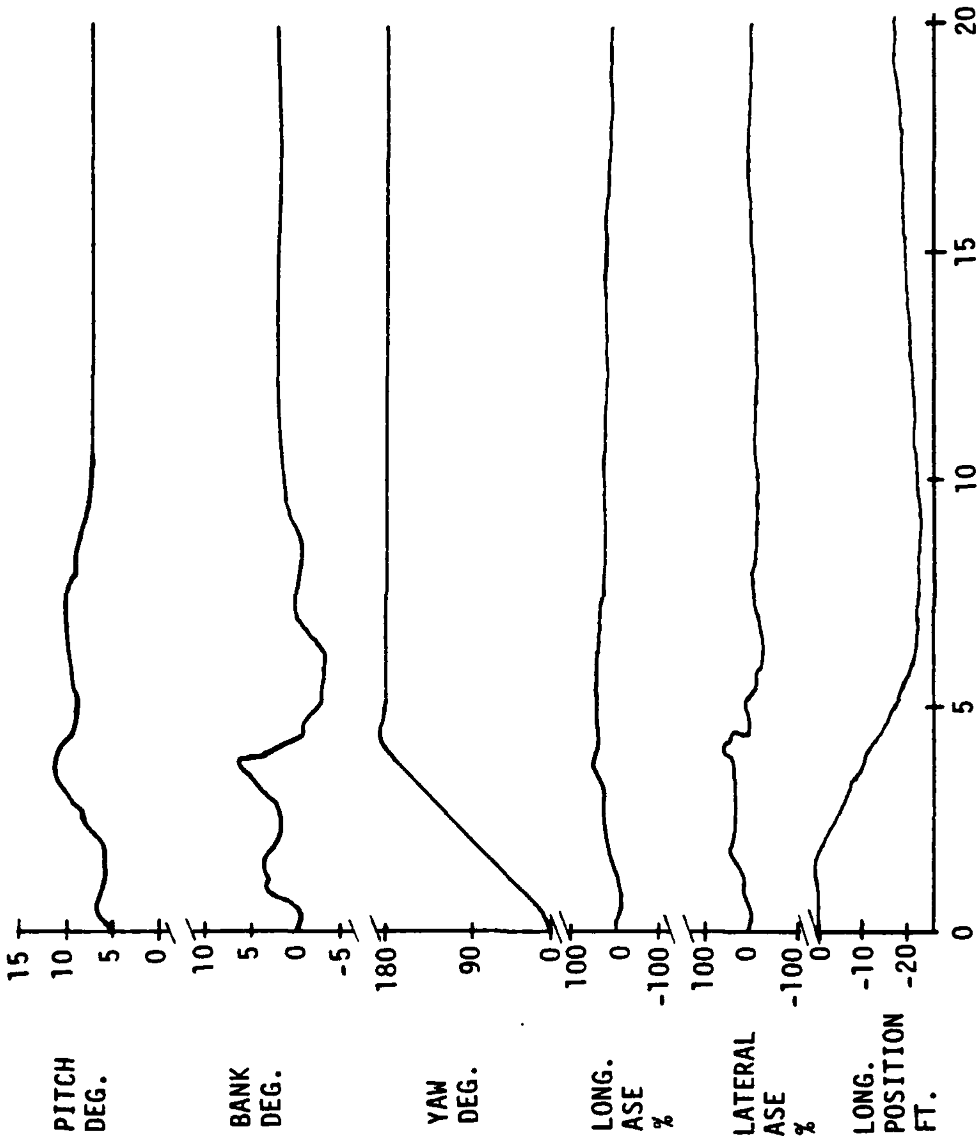

Figure 21. SH-2F Model Response to a 180 Degree Turn Over a Spot with 35 knot wind, position and integral feedback

7. Turn Over a Spot

Paragraph 3.3.6 "It shall be possible to execute a complete turn in each direction while hovering over a given spot at the maximum overload gross weight or at takeoff power (in and out of ground effect), in a wind of at least 35 knots. To insure adequate margin of control during these maneuvers, sufficient control shall remain at the most critical azimuth angle relative to the wind, in order that, when starting at zero yawing velocity at this angle, the rapid application of full directional control in the critical direction results in a corresponding yaw displacement of at least $\sqrt[3]{\frac{110}{W + 1000}}$ degrees in the first second, where W represents the maximum overload gross weight of the helicopter in pounds."

The ground speed hold mode of the SH-2F was modified to perform a turn over a spot maneuver. It was necessary to increase the A.S.E. control authority to 100 percent and to add a position hold loop. Ground position error was resolved into body axis components to determine proper pitch and bank commands. An integral of airspeed error feedback was added to the longitudinal stick command to cancel out trim changes and minimize steady state position error. The model was trimmed in a ground referenced hover with 35 knots wind at zero sideslip angle. The aircraft was then directed to make a 60 degree/second turn to the right. Both 180 degree and 360 degree turns were simulated. Figure 21 shows the 180 degree turn and figure 22 shows the 360 degree turn. In both cases, the yaw response is well damped and closely follows the commanded heading angle. Longitudinal and lateral stick commanded during the 180 degree turn remain well within control limits. Lateral position error remains very small, but the aircraft tends to drift down wind during the turn. The integral compensation attempts to restore the aircraft to the initial hover point, but the response is sluggish. Aircraft response to the 360 degree turn is similar to that of the 180 degree turn because the aircraft returns to its initial trim condition. Longitudinal and lateral control displacement remain well within available limits for the 360 degree turn.

8. Control Power Available at Critical Azimuth

The SH-2F model was trimmed with a 35 knot airspeed at sideslip angles ranging from 0 to 330 degrees. Figure 23 summarizes the trim characteristics of the model. The plot shows that adequate control power remains with the aircraft trimmed with 35 knot wind from any direction including left, right, and rearward flight.

The sideslip sweep was repeated for a speed of 15 knots and the results were compared with flight test data. Figure 24 shows this comparison. The collective and directional controls show a very close match over most of the sideslip range. However, the model underestimates the lateral stick required to trim at large slideslip angles. Longitudinal stick trim shows a reasonable comparsion for moderate sideslip angles, but the aircraft requires more aft stick to trim for large sideslip angles than does the model.

9. Direction Stability

Paragraph 3.3.9 "The helicopter shall possess positive, control fixed, directional stability, and effective dihedral in both powered and autorotative flight at all forward speeds above 50 knots, 0.5 V_{max}, or the speed for maximum rate of climb, whichever is the lowest. At these flight conditions with zero yawing and rolling velocity, the variations of pedal displacement and lateral control

Table III

Comparison of Estimated Control Power and Damping with Required Control Power and Damping

Aircraft	M_δ		M_q		L_δ		L_p		N_δ		N_r	
	Est.	Req.	Est.	Req.	Est.	Req.	Est.	Req.	Est.	Req.	Est.	Req.
OH-6A	.7408	.277	-1.76	-1.95	1.279	.572	-4.9	-4.39	2.5	.329	-.86	-3.7
BO-105	.9727	.3398	-3.39	-1.285	2.64	.805	-9.24	-2.88	1.39	.2401	-.32	-2.3
AH-1G	.1462	.1322	-.2345	-.878	.4709	.2429	-.766	-2.336	.81	.22	-.5	-1.6
UH-1H	.169	.1304	-.19	-.92	.56	.235	-.57	-2.28	1.19	.23	-.7	-1.7
H-53D	.179	.091	-.499	-.39	.515	.170	-1.86	-1.07	.35	.13	-.36	-.716
SH-2F	.39778	.2522	-3.86	-.81	.308	.430	-2.029	-1.8	1.52	.44	-2.135	-1.5
SH-60B	.388	.107	-.562	-.59	1.467	.26	-3.6	-1.74	.69	.193	-.5	-1.09

For a given value of damping and a required attitude response after a time Δt, the required control power may be calculated as

$$M_\delta = \frac{\Delta \theta (t)}{\frac{-\Delta t}{Mq} + \frac{1}{(Mq)^2} \cdot (e^{Mq} \cdot \Delta t_{-1})}$$

$$L_\delta = \frac{\Delta \phi (\Delta t)}{\frac{-\Delta t}{L_p} + \frac{1}{(L_p)^2} \cdot (e^{L_p} \cdot \Delta t_{-1})}$$

$$N_\delta = \frac{\Delta \psi (\Delta t)}{\frac{-\Delta t}{N_r} + \frac{1}{(N_r)^2} \cdot (e^{N_r} \cdot \Delta t_{-1})}$$

Required damping was calculated for available helicopters based on the specified values contained in 3.6.1.1. These were compared with estimated values of actual damping. Required attitude response was calculated for each helicopter based on gross weight. Finally, the required control power was calculated based on desired attitude response and actual damping. Table III compares actual control power and damping with required control power and damping.

Several trends are apparent from the data tabulated in Table III. Both rigid rotor helicopters (OH-6A and BO-105) substantially exceed the required pitch and roll control power and meet or exceed the required pitch and roll damping. At the other extreme in rotor design, both teetering rotor helicopters (AH-1G/UH-1N) just meet the pitch and roll control power requirements, but fall short of the damping requirements. The low available damping partially compensates for the low control power enabling these helicopters to be reasonably maneuverable. The control power and damping of the three hinged rotor helicopters (CH-53D, SH-2F, SH-60B) falls between the teetering rotor and rigid rotor data. Yaw control power substantially exceeds the required values for all the single rotor helicopters, and damping falls short of the desired values for all helicopters except the SH-2F.

Estimated derivatives for the SH-2F appear to correlate with the time history calculations for the model. Both the derivative values and time history calculations indicate that the model has low roll control power and high yaw damping when compared with both flight test data for the SH-2F and also comparison with other similar helicopters. Very limited pilot opinion data is available to determine the adequacy of the control and damping requirements. In addition, with the exception of the OH-6A and BO-105, the remaining helicopters are normally flown with augmentation systems engaged. MIL-H-8501A does not distinguish requirements for A.S.E. on and A.S.E. off damping or control power. By the nature of their design single rotor helicopters are more limited by yaw trim constraints than by yaw maneuvering control power. Thus, yaw stability augmentation is usually required to compensate for low damping.

For adequate flying qualities, there must be a proper balance between control power and damping. The current specification separately specifies minimum requirements for each, but places no restriction on the ratios of control power to damping. Only the yawing axis has an upper limit on attitude response. An upper bound may be needed for pitch and roll as well if rigid rotor designs become more common.

TABLE II

SH-2F Model Attitude Response in Hover
vs.
Flight Test Data and MIL-H-8501A Requirements

	Model	Flight Test	MIL-H-8501A VFR	MIL-H-8501A IFR
Pitch attitude in 1 sec. with 1 inch stick -deg.	$+3.6^\circ$ -3.46°	$+4^\circ$ -8°	$\pm 1.87^\circ$	$\pm 3.0^\circ$
Pitch attitude in 1 sec. with full stick -deg.	$+18.37^\circ$ -23.51°	-	$\pm 7.48^\circ$	$\pm 12^\circ$
Roll attitude in 0.5 sec. with 1 inch stick -deg.	$+1.21^\circ$ -1.21°	$+2^\circ$ -2°	$\pm 1.1^\circ$	$\pm 1.3^\circ$
Roll attitude in .5 sec. with full stick -deg.	$+7.24^\circ$ -5.27°	-	$\pm 3.37^\circ$	$\pm 4.5^\circ$
Yaw attitude in 1 sec. with 1 inch pedal -deg.	$+21.53^\circ$ -20.01°	$+44^\circ$ -45°	$\pm 4.5^\circ$	$\pm 4.5^\circ$
Yaw attitude in 1 sec. with full pedal -deg.	$+48.66^\circ$ -49.45°	-	$\pm 13.5^\circ$	$\pm 13.5^\circ$

A simplified linear analysis was performed to obtain a relationship between control power, damping, and attitude response in one second. First order response was assumed for pitch, roll, and yaw response to longitudinal, lateral, and rudder controls respectively. The simplified attitude equations then become

$$\ddot{\theta} = M_\delta \cdot \delta + M_q \cdot q \text{ - rad/sec}^2$$

$$\ddot{\phi} = L_\delta \cdot \delta + L_p \cdot p \text{ - rad/sec}^2$$

$$\ddot{\psi} = N_\delta \cdot \delta + N_r \cdot r \text{ - rad/sec}^2$$

Paragraph 3.6.1.1 For any helicopter required to operate under instrument or all-weather conditions, the following control power and angular velocity damping requirements shall apply in hovering:

	Angular displacement at end of 1 sec. for a rapid 1-inch control displacement-degrees	Angular velocity damping ft-lbs/ rad/sec.
Longitudinal	$\sqrt[3]{\frac{73}{W + 1000}}$	15 (I_y) 0.7
Directional	$\sqrt[3]{\frac{110}{W + 1000}}$	27 (Iz) 0.7
Lateral	(1)	25 (Ix) 0.7

[1] The lateral requirement is based on the angular displacement at the end of one-half second following a control displacement and for a 1-inch control displacement shall be at least $\sqrt[3]{\frac{32}{W + 1000}}$ degrees displacement in the first one-half second. For full available displacement of the controls from trim, the values of angular displacement specified above shall be multiplied by four for longitudinal and three for lateral and for directional values."

Pitch, roll, and yaw responses to control step inputs were calculated for the SH-2F model trimmed in hover at a maximum allowed weight of 12,800 pounds under standard day sea level conditions. Pitch responses were determined for one-inch forward and aft stick inputs and for full forward and aft stick inputs. As shown in Table II, the model noseup pitch response compares well with flight test. However, the flight test data indicates a more rapid nosedown response than is indicated by the model. Both the model and the flight test results exceed IFR specification requirements. Model pitch response for full control authority significantly exceeds requirements.

The model roll response to one-inch lateral stick meets specification requirements, but calculated bank is substantially less than measured flight response. This indicates that further refinement of the model roll response may be required. Measurement of bank angle in 0.5 seconds is very difficult because it strongly depends on obtaining a precise trim prior to the maneuver and on the shape of the control input. Testing procedures would be simplified if a one-second criteria were adopted that would be consistent with the one-second pitch and yaw criteria.

Calculated yaw response is significantly less than the measured attitude change. The model yaw damping was increased based on pilot comments. Additional test data is needed to determine the actual helicopter characteristics and to refine the model. Both the model and flight yaw characteristics indicate a very high yaw response that may be unsatisfactory. The flight response may be excessive because it approaches the recommended maximum response of 50 degrees in one second. However, the aircraft is normally operated with yaw ASE engaged so this may not be a significant problem.

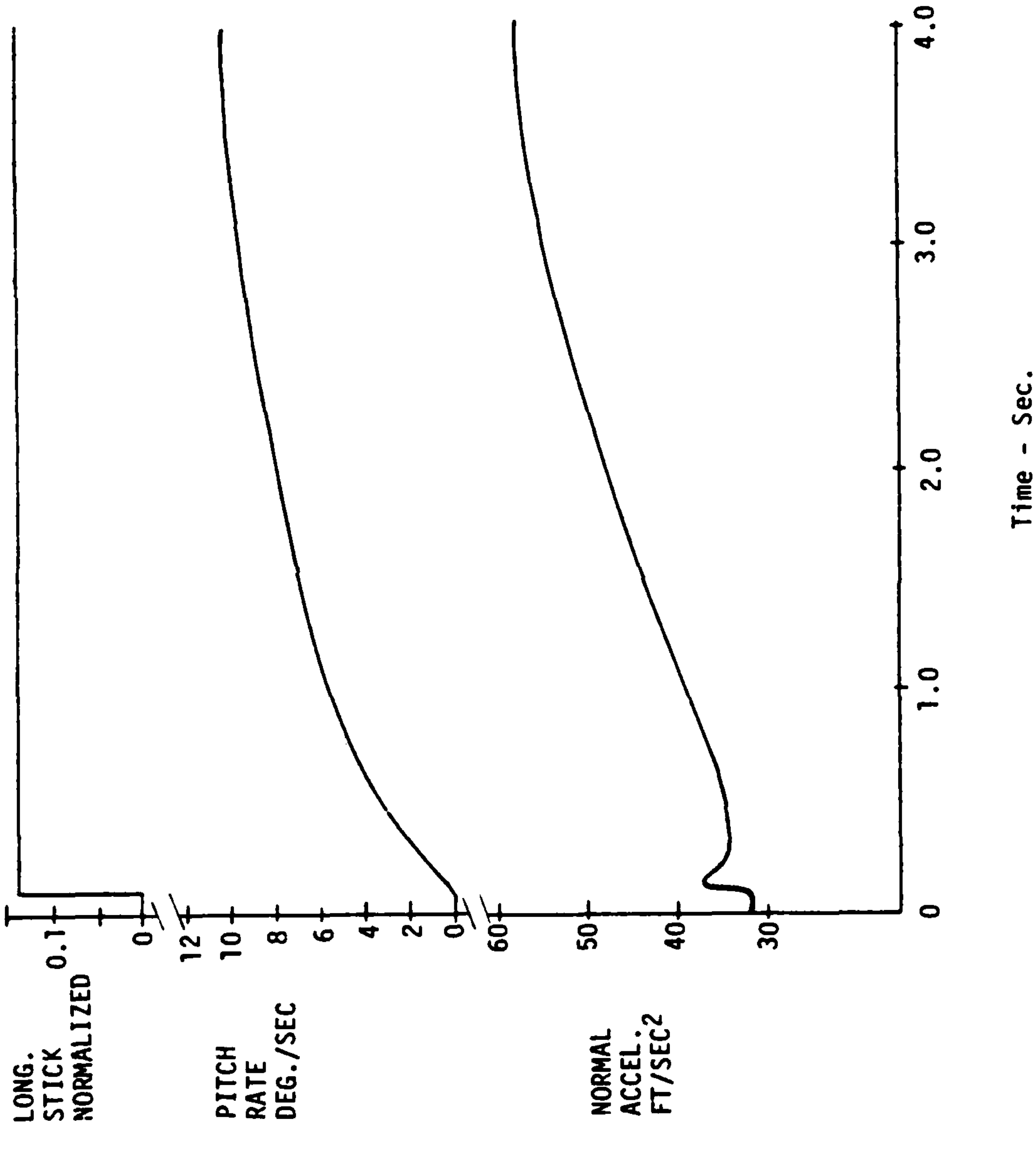

Figure 20. SH-2F Model Response to 1 inch aft stick at 120 knots

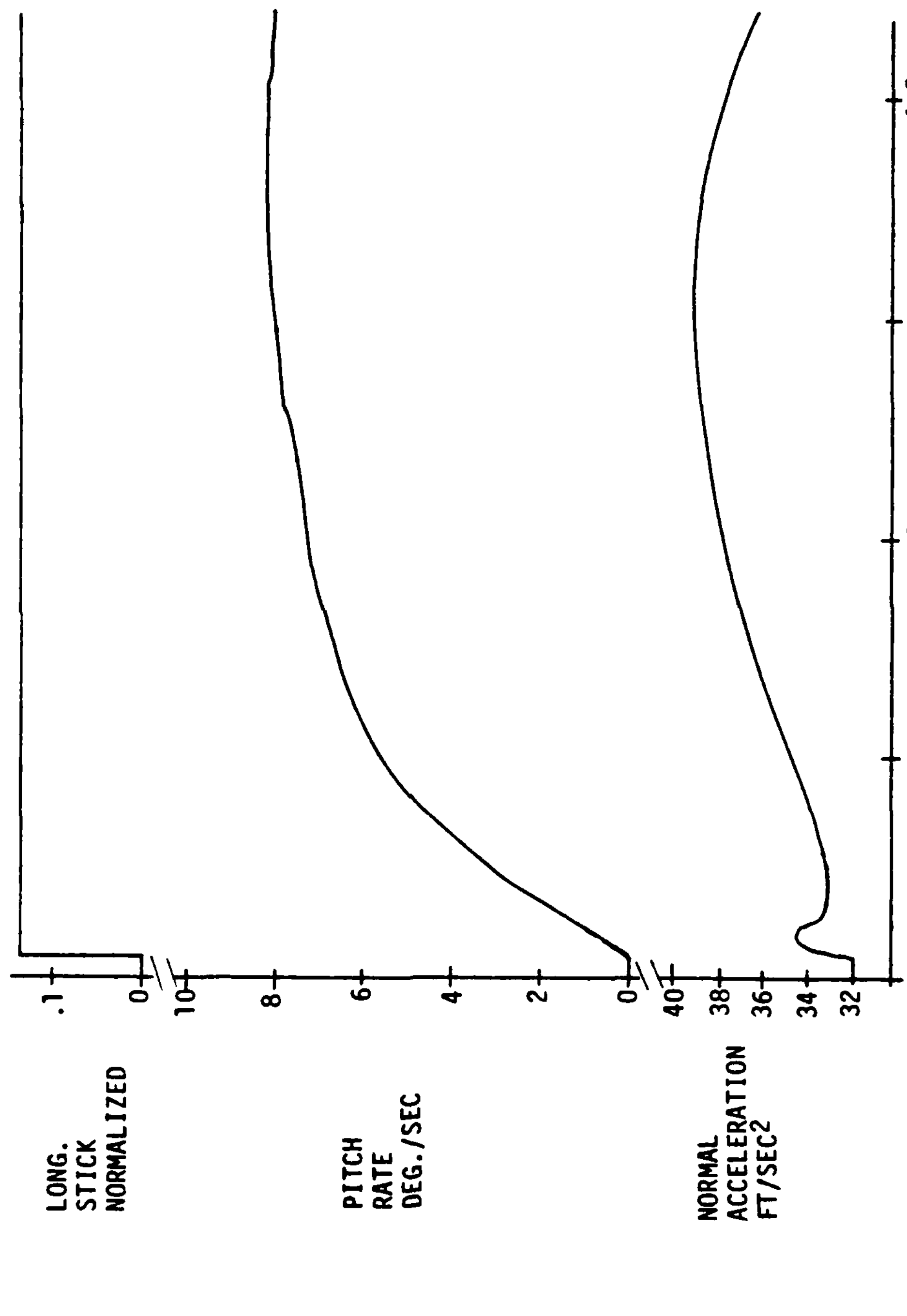

Figure 19. SH-2F Model Response to 1 inch stick at 60 knots

The calculation is very straight forward for the model data because exact numerical values of attitude and pitch rate are available. As shown in figure 19 at 60 knots, the one-inch stick input limit again determined the size of the maneuver. The peak pitch rate was 8.3 degrees/second with the pitch rate inflection point occurring 0.6 seconds after the control input. A peak normal acceleration of 1.22 g's is achieved three seconds after the step input, and the normal acceleration becomes concave downward after 0.9 seconds.

For the 120 knot trim shown in figure 20, the one-inch stick input causes the normal acceleration to exceed 1.5 g's after three seconds. Nevertheless, the normal acceleration becomes concave downward after 0.8 seconds and the pitch rate becomes concave downward after 0.4 seconds. In all cases the model responses become concave downward well within the two-second limit required by MIL-H-8501A. Unless flight test data were relatively noise free, it would be difficult to determine the compliance of the helicopter with this requirement. Overall, the requirement appears very lenient and imposes few restrictions on the response of even very unstable helicopters. This portion of the specification could probably be eliminated if the longitudinal dynamic stability requirement were better defined.

6. Attitude Response in Hover

Paragraph 3.2.13: "Longitudinal control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1.0-inch step displacement from trim of the longitudinal control shall produce an angular displacement at the end of 1.0 second which is at least $\frac{45}{\sqrt[3]{W + 1000}}$ degrees. When maximum available displacement from trim of the longitudinal control is rapidly applied, the angular displacement at the end of 1.0 second shall be at least $\frac{180}{\sqrt[3]{W + 1000}}$ degrees. In both expressions W represents the maximum overload gross weight of the helicopter in pounds.

Paragraph 3.3.5 Directional control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at rated takeoff power, a rapid 1.0-inch step displacement from trim of the directional controll shall produce a yaw displacement at the end of 1.0 second which is at least $\frac{110}{\sqrt[3]{W + 1000}}$ degrees. When maximum available displacement from trim of the directional control is rapidly applied at the conditions specified above, the yaw angular displacement at the end of 1.0 second shall be at least $\frac{330}{\sqrt[3]{W + 1000}}$ degrees. In both equations W represents the maximum overload gross weight of the helicopter in pounds.

Paragraph 3.3.18 Lateral control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid one-inch step displacement from trim of the lateral control shall produce an angular displacement at the end of one-half second of at least $\frac{27}{\sqrt[3]{W + 1000}}$ degrees. When maximum available displacement from trim of the lateral control is rapidly applied at the conditions specified above, the resulting angular displacement at the end of one-half second shall be at least $\frac{81}{\sqrt[3]{W + 1000}}$ degrees. In both expressions W represents the maximum overload gross weight of the helicopter in pounds.

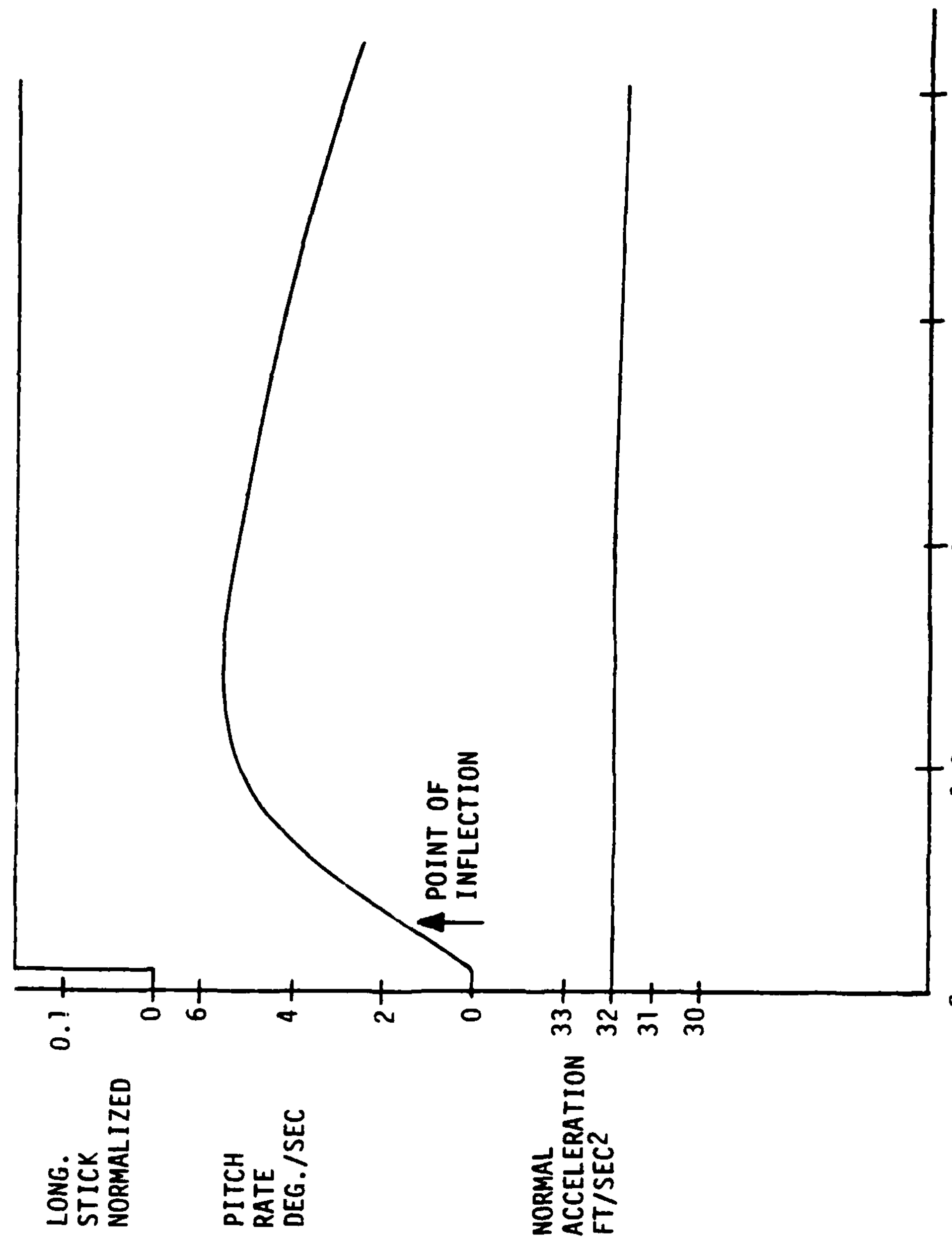

Figure 18. SH-2F Model Response to 1 inch aft stick in hover

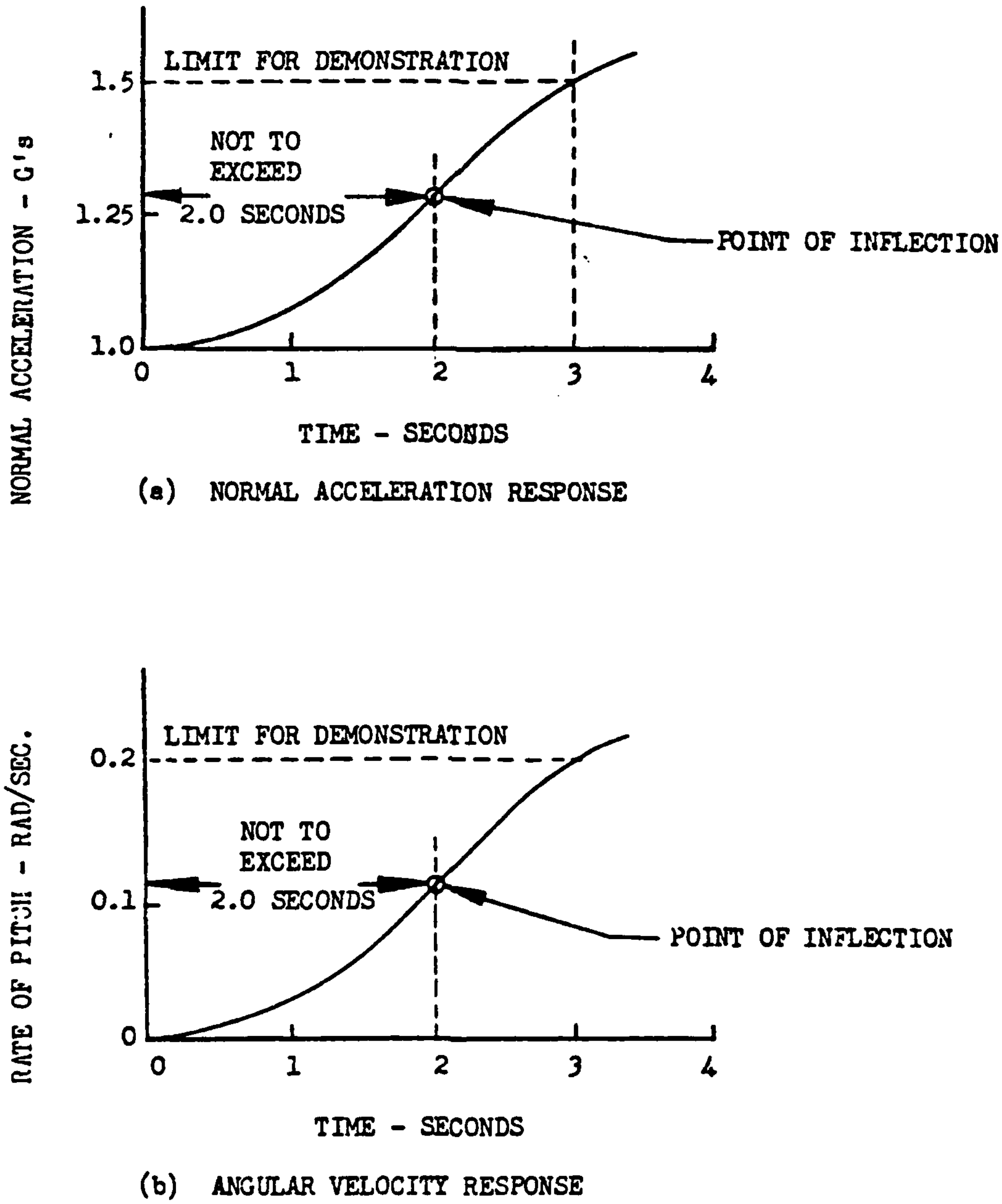

Figure 17. Typical Normal Acceleration and Pitch Rate Response

displacement with steady sideslip angle shall be stable (left pedal and right stick displacement for right sideslip) up to full pedal displacement in both directions, but not necessarily beyond a sideslip angle of 15 degrees at V_{max}, 45 degrees at the low speed determined above, or beyond a sideslip angle determined by a linear variation with speed between these two angles. Between sideslip angles of ± 15 degrees, the curve of pedal displacement and lateral control displacement plotted against sideslip angle shall be approximately linear. In all flight conditions specified above, a ten percent margin of both lateral and longitudinal control effectiveness (as defined in 3.2.1 and 3.3.4) shall remain."

Roll and yawing moment variation with side velocity were calculated for a range of speeds from 0 to 90 knots for level flight and for autorotation at 80 knots. These results were compared with the specification requirements. MIL-H-8501A requires positive directional stability and effective dihedral above 50 knots. Table IV summarizes the characteristics of the model.

Table IV

SH-2F Model Static Directional Stability
vs.
Forward Speed

Speed — knots	0	35	60	80	80	90
Flight Condition	level flt.	level flight	level flight	auto-rotation	level flight	level flight
N_V	.0415	.03152	.0369	.0416	.04414	.04779
L_V	-.02812	-.0166	-.0171	-.01216	-.01956	-.02097

The model demonstrated positive directional stab‥.y and stable dihedral effect over the entire speed range investigated including hover. Only the taii rotor generates stabilizing moments in hover. As speed increases, the contribution of the fueslage and vertical tail becomes more important. Dihedral effect is nearly constant above 35 knots whereas directional stability increases somewhat with speed. Dihedral effectiveness is reduced as the model transitions from level flight to autorotation at 80 knots airspeed. Root locus analysis of the two model configurations indicate the presence of a divergent mode with a time to double amplitude that decreases from 17 seconds to 6.6 seconds as the sink rate increases from 0 to 40 feet/second with a forward speed of 80 knots. However, the divergent mode appears to be primarily associated with the longitudinal dynamics of the model. The reduction in dihedral effect does not significantly affect the characteristics of the model.

Flight test results of the SH-2D described in reference (g) indicate excessive collective to longitudinal coupling in autorotation and marginal directional control power. However, no difficulty was reported in maintaining autorotation above 60 knots. The flight test data suggest the existence of longitudinal control deficiencies as predicted by the analytic model.

10. Autorotation Characteristics

Section 3.5 of MIL-H-8501A specifies helicopter autorotation requirements as follows:

Paragraph 3.5.5. "The helicopter shall be capable of entering into power-off autorotation at all speeds from hover to maximum forward speed. The transition from powered flight to autorotative flight shall be established smoothly, with adequate controllability and with a minimum loss of altitude. It shall be possible to make this transition safely when initiation of the necessary manual collective-pitch control motion has been delayed for at least two seconds following loss of power. At no time during this maneuver shall the rotor speed fall below a safe minimum transient autorotative value (as distinct from power-on or steady-state autorotative values). This shall be construed to cover both single and multiengine helicopters.

Paragraph 3.5.5.1 Sudden power reduction, power application, or loss of power with collective control fixed, shall not produce pitch, roll, or yaw attitude changes in excess of ten degrees in two seconds, except that, at speeds below that for best climb, a 20-degree yaw in two seconds will be accepted."

A dynamic simulation of engine failure and autorotation entry was performed using the SH-2F model. Engine failure was represented by setting engine drive torque to zero at a specified time. Pilot control response was approximated by lowering collective to full down at a prescribed delay time after the engine failure. The attitude control system was engaged to maintain initial trim pitch, roll, and heading attitude. Helicopter response was simulated starting from level flight at speeds of 60, 80, and 100 knots. A one-second delay between engine failure and collective reduction was selected to represent pilot reaction time. Figures 25-30 summarize the calculated response characteristics of the model.

In all three cases, rotor speed decreases rapidly following simulated engine failure at Time = 1 second. Sink rate builds up rapidly beginning at Time = 2 seconds when the collective is lowered. A steady sink rate is achieved within seven - ten seconds and the rotor speed begins to recover about three seconds after lowering collective. A tendency toward overspeeding the rotor is evident as the initial forward speed is increased. A more sophisticated control strategy is needed to increase the collective until the desired rotor speed is achieved.

Pitch and roll coupling to collective control increase with speed. At speeds above 80 knots, the simulated attitude control system is unable to maintain wings level flight in autorotation. This results partially from deficiencies in the model, but it is also indicative of difficulties in the helicopter as reported in reference (g). Additional flight test data are needed both to validate the model characteristics and to establish pilot handling qualities rating for the helicopter. Based on the calculated rotor speed loss after one second, it is unlikely that a recovery could be made if pilot collective input were delayed for two seconds after power loss. However, the instantaneous loss of all engine power is probably an unrealistically extreme test.

11. Attitude Response Following Sudden Loss of Power

The effect of a sudden power failure with controls fixed was simulated to examine the requirements of 3.5.5.1. Figures 31-34 show the simulated attitude response for initial speeds of 0, 60, and 120 knots. Engine failure was simulated one second after the start of the maneuver and the controls were held fixed for the duration of the maneuver. Pitch and bank response met or only slightly exceeded the ten degree excursion limit at all speeds examined. However, heading change greatly exceeded the allowable limit in hover. The specification is probably unrealistically strict in hover because available yawing moment is low. At higher speeds, the simulated yawing response complies with the specification. No comparable flight test data is available to validate the model or to determine the validity of the requirement.

The most pronounced deficiency in the model response appears to be the extremely rapid rotor deceleration. Rotor speed falls to 80 percent of its initial value within two seconds of the engine failure. It is doubtful if a successful recovery could be made starting from this point. Simulated rotor speed change is seven percent in the first 0.5 seconds after engine failure compared to a

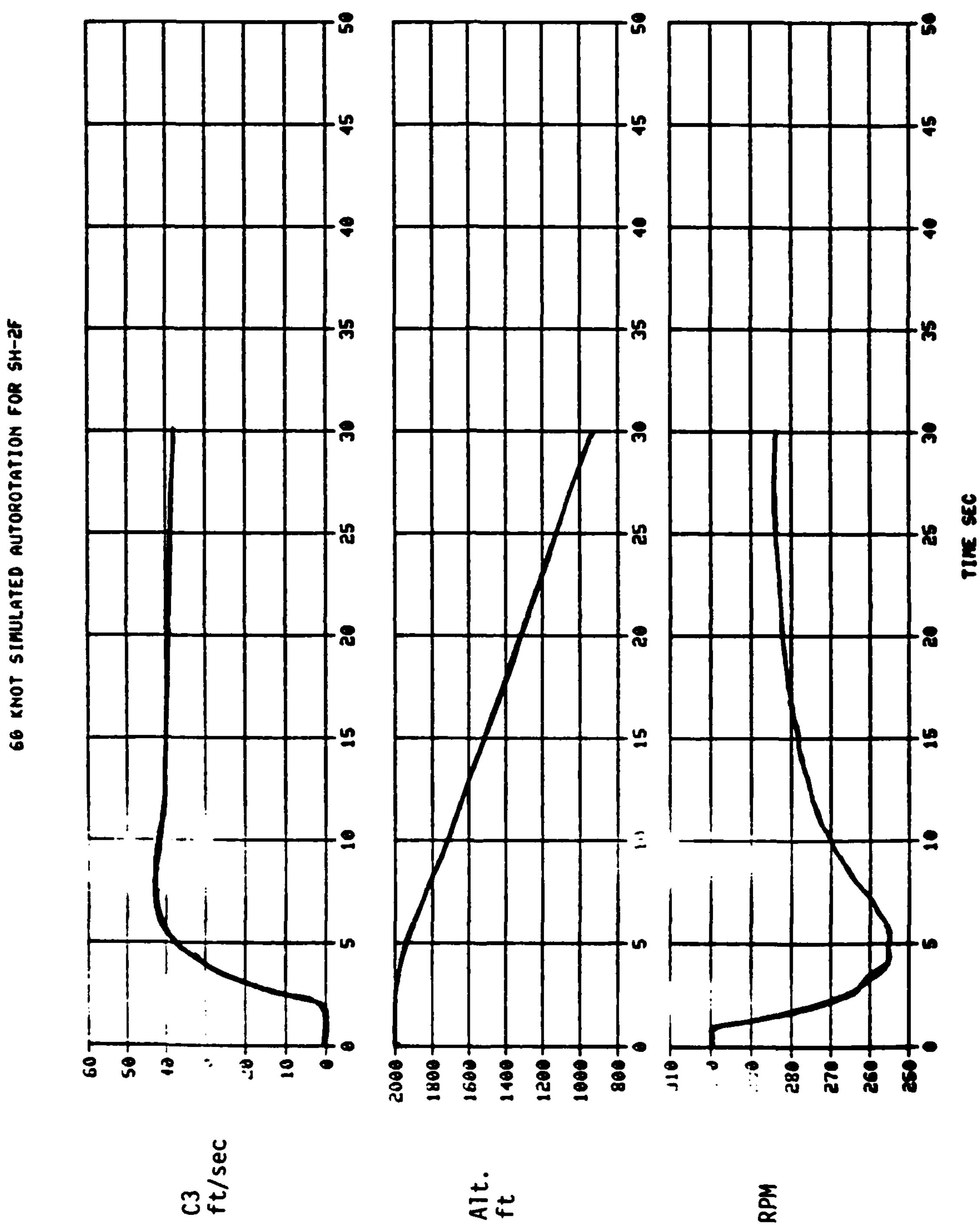

Figure 25. 60 Knot Simulated Autorotation for SH-2F

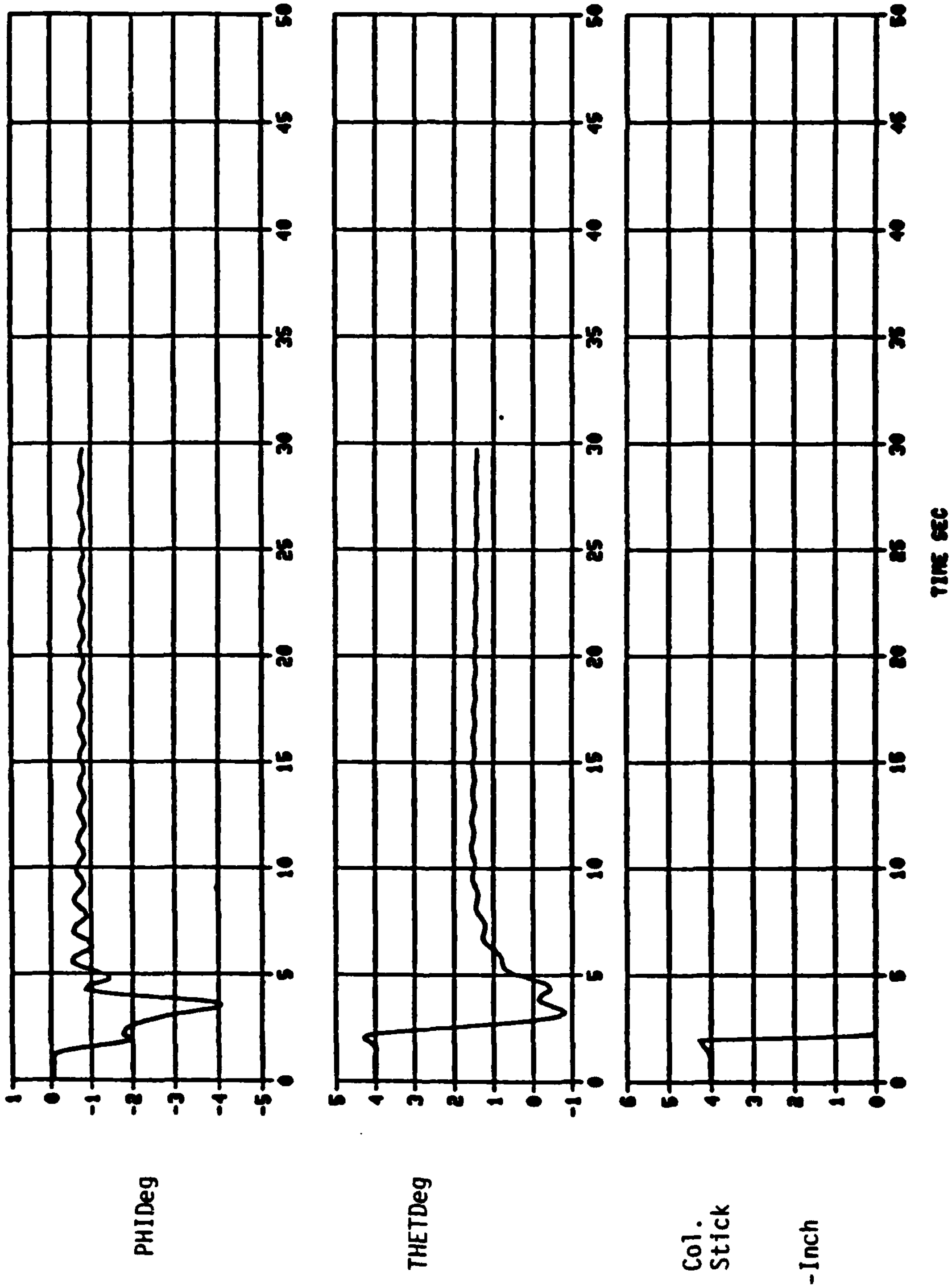

Figure 26. 60 Knot Simulation for SH-2F, Attitude Response

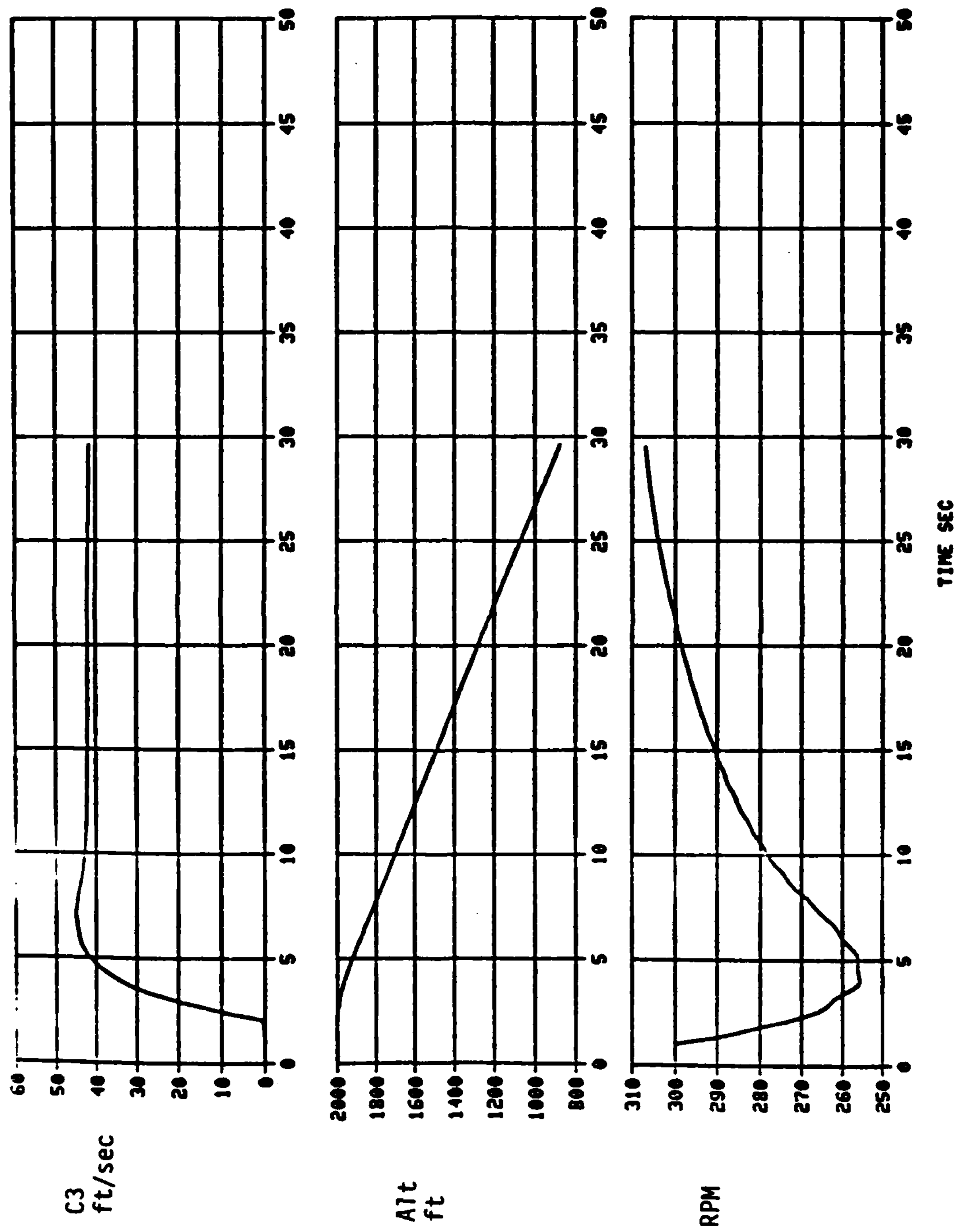

Figure 27. Simulated SH-2F Autorotation, V = 80 knots

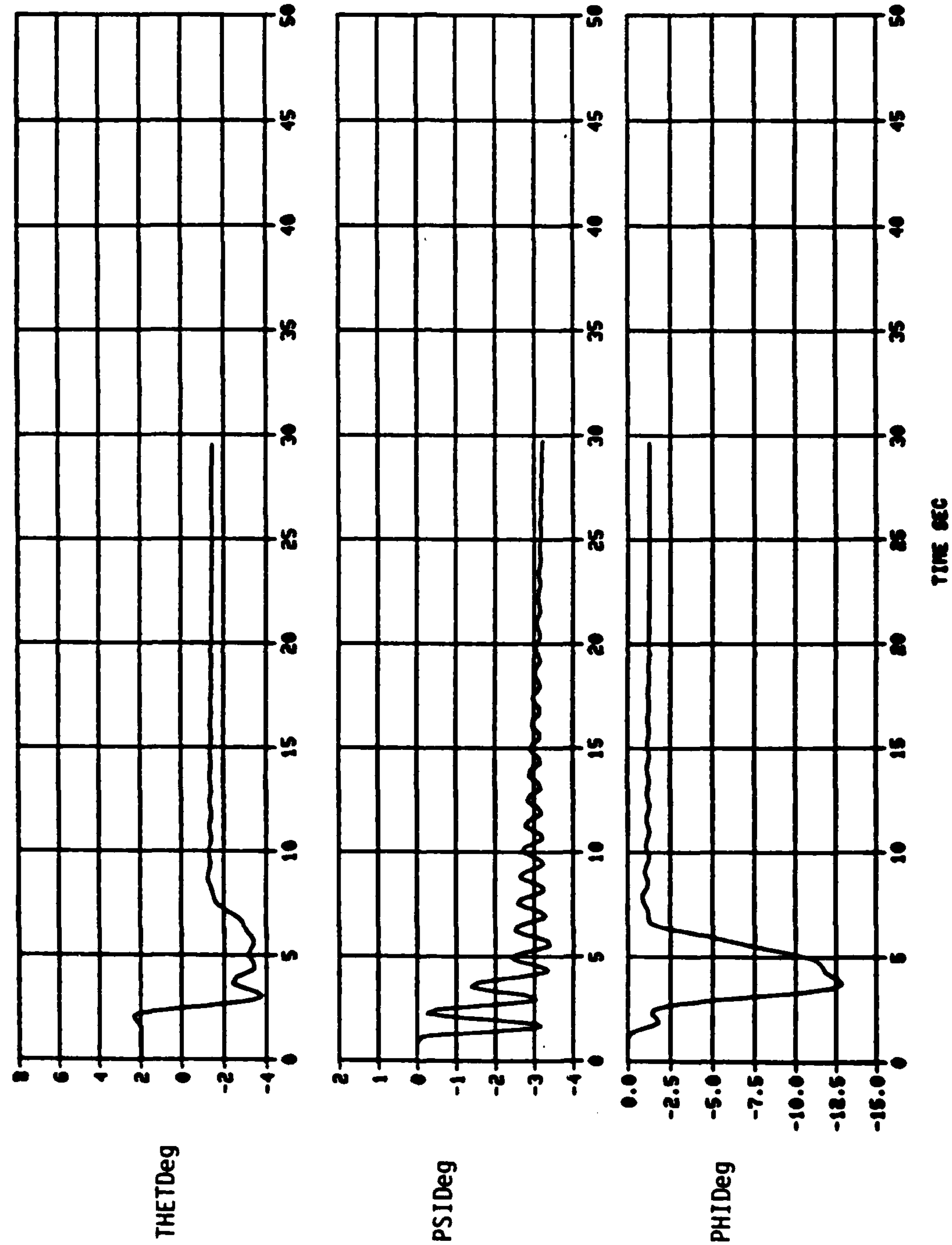

Figure 28. Simulated SH-2F Autorotation, V = 80 knots, Attitude Response

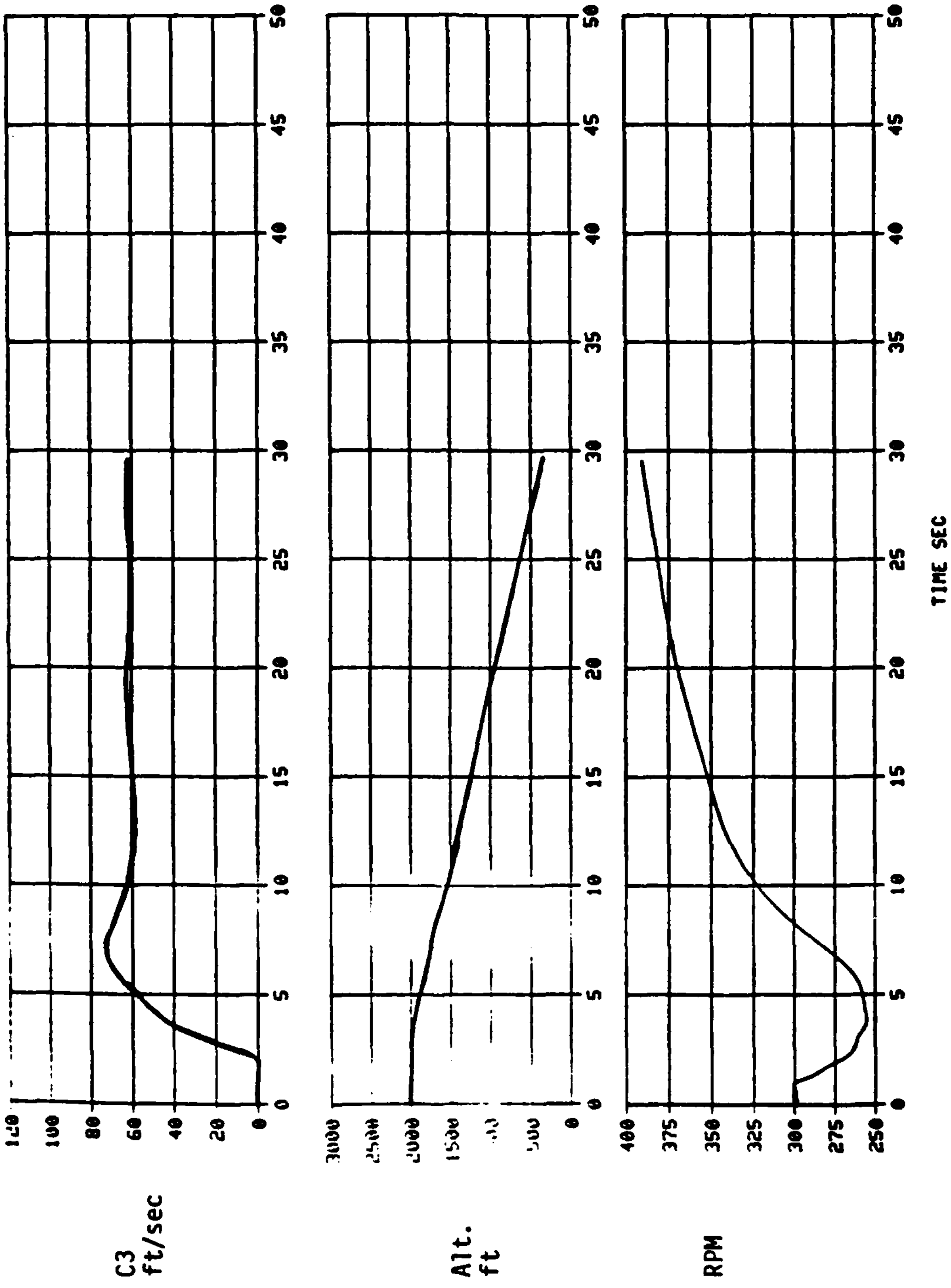

Figure 29. Simulated SH-2F Autorotation, V = 100 knots

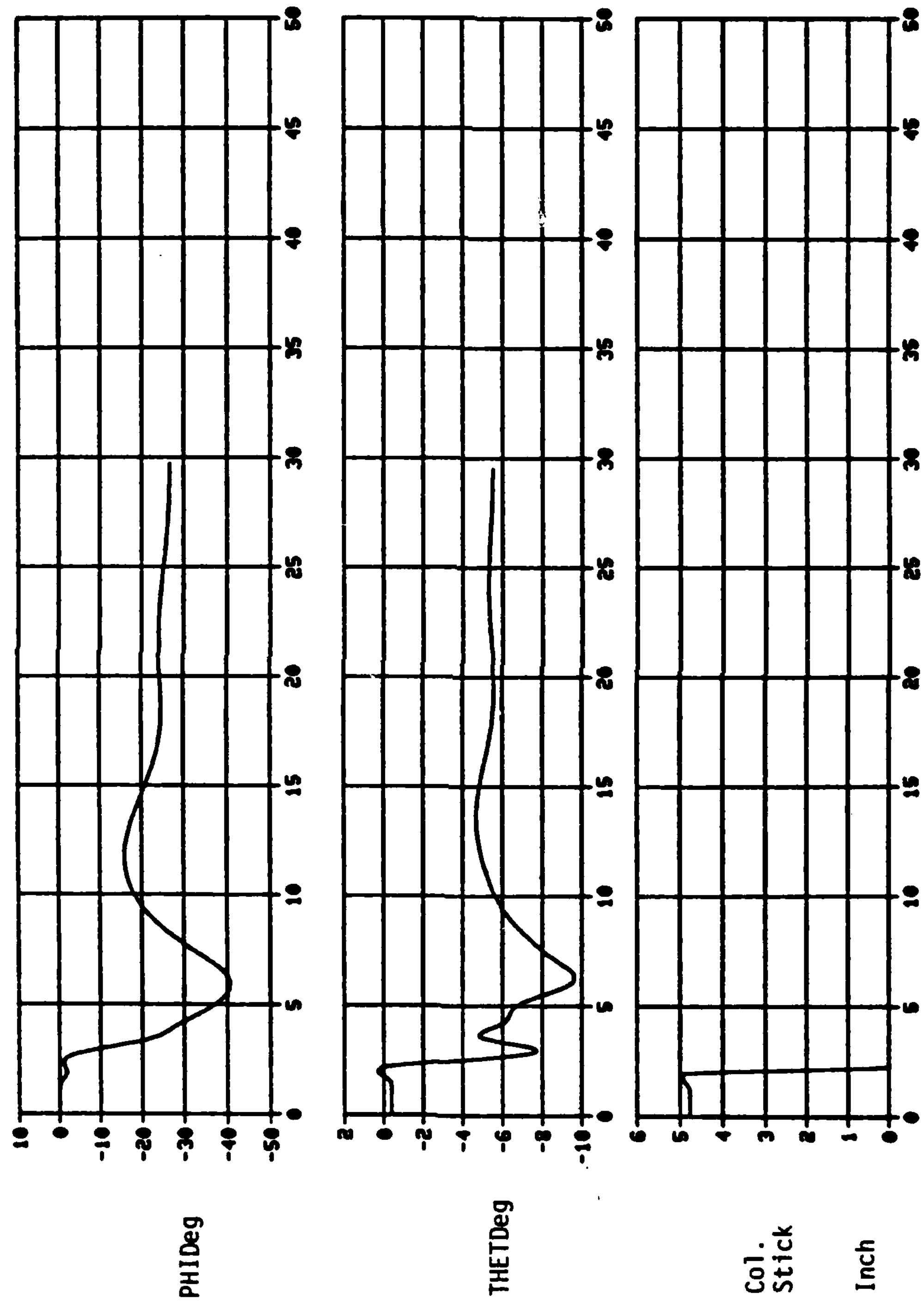

Figure 30. Simulated SH-2F Autorotation, V = 100 knots, Attitude Response

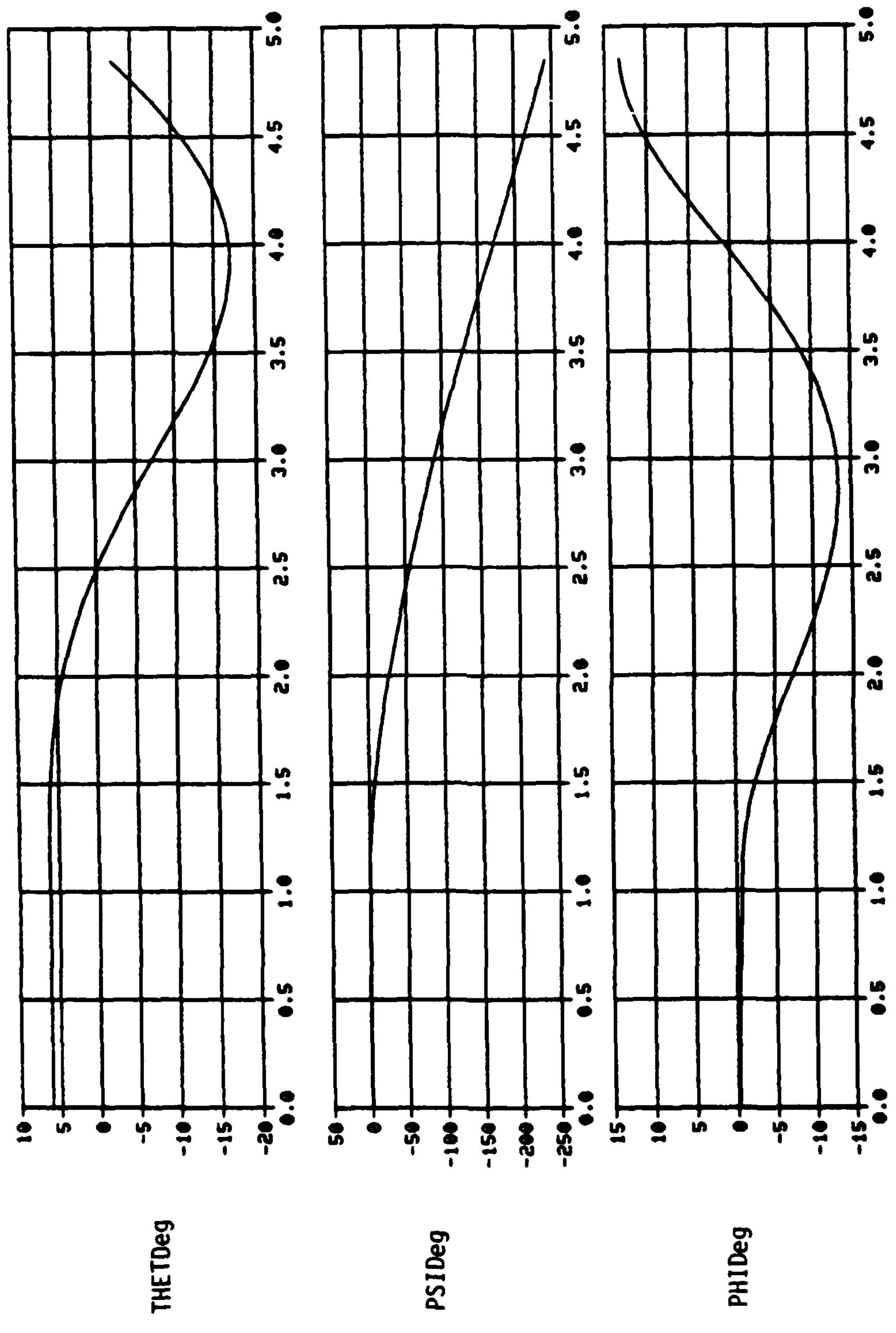

Figure 31. Simulated Engine Failure with Fixed Controls, V = 0

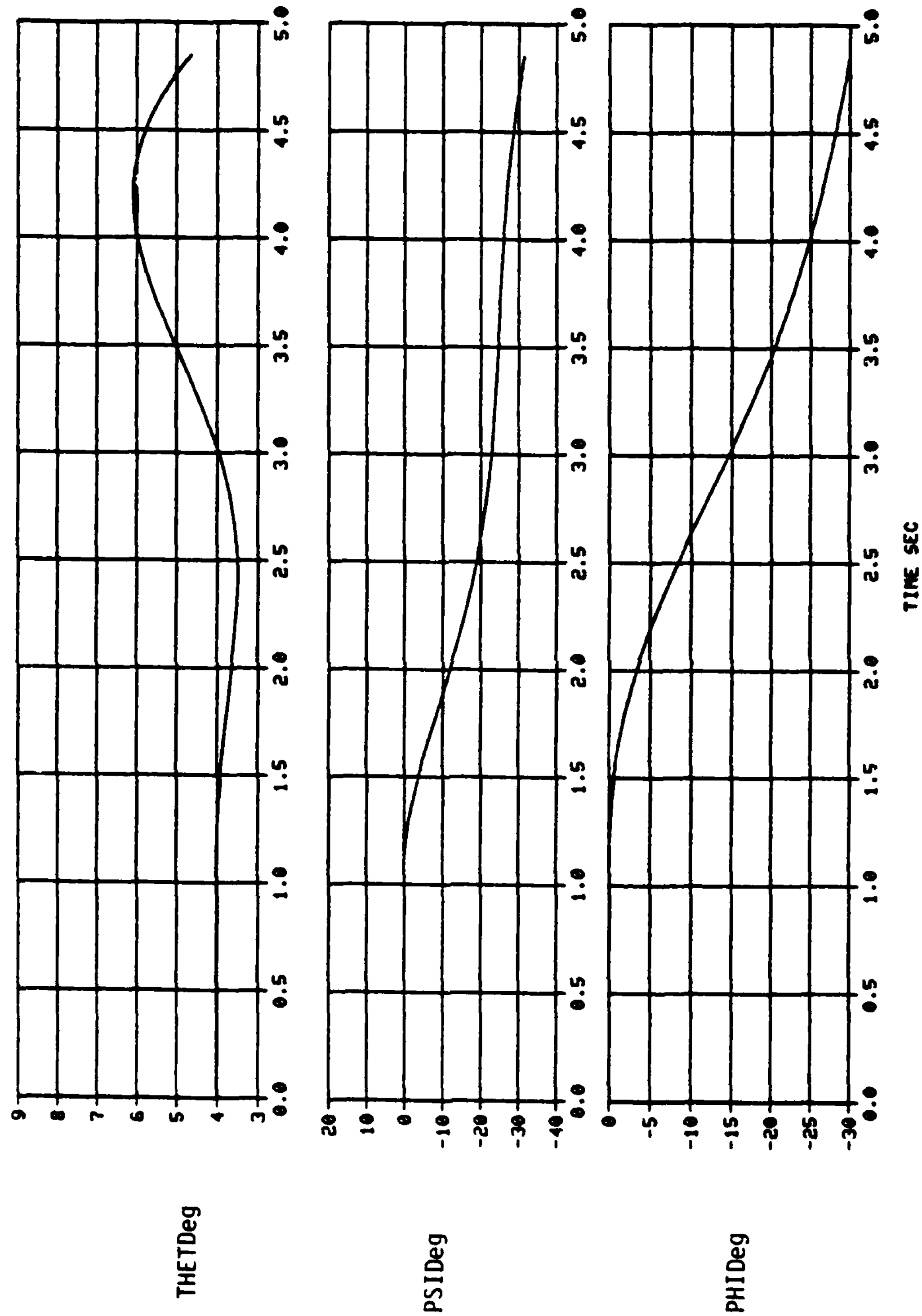

Figure 32. Simulated Engine Failure with Controls Fixed, V = 60 knots

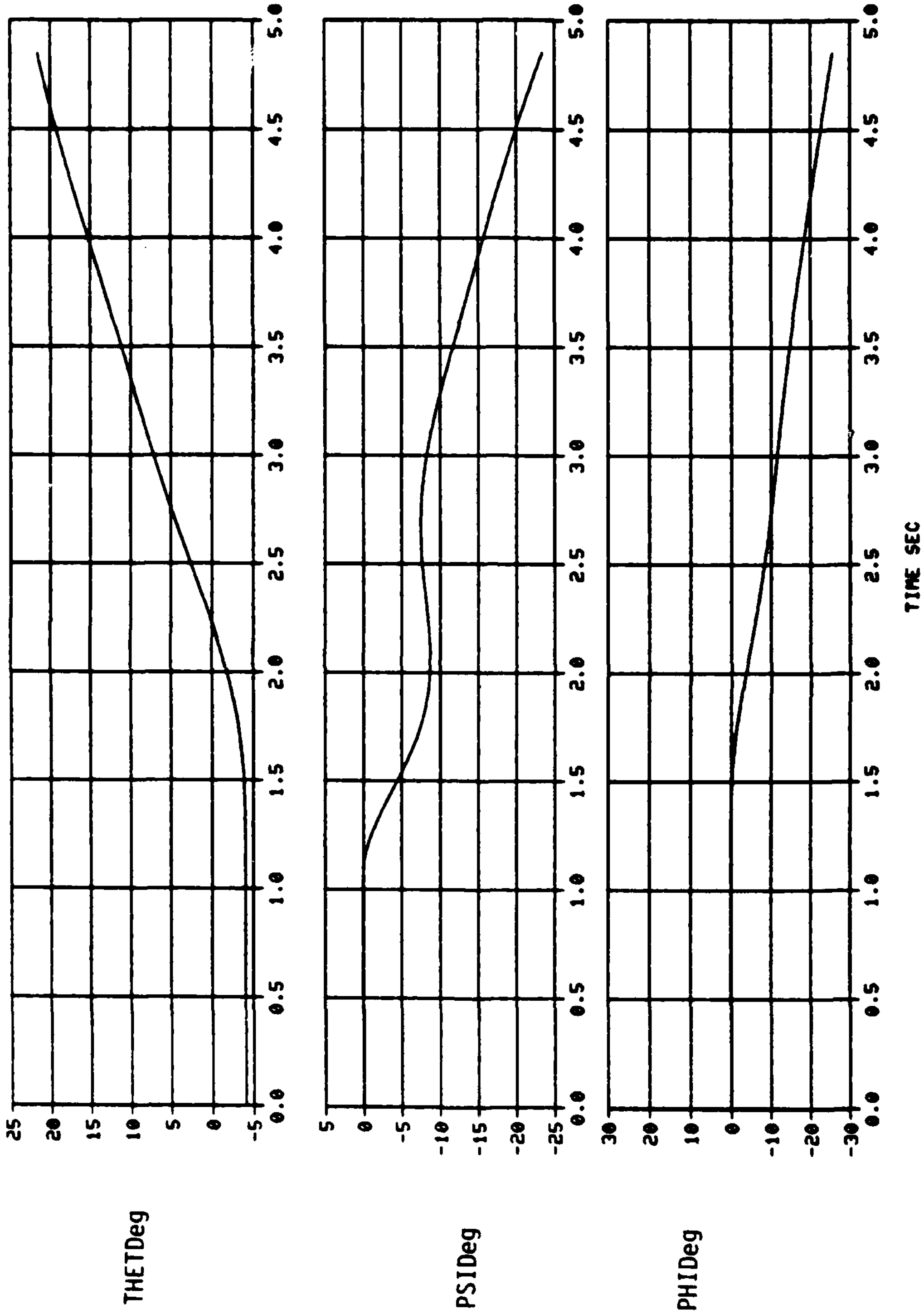

Figure 33. Simulated Engine Failure with Controls Fixed, V = 120 knots, Attitude Response

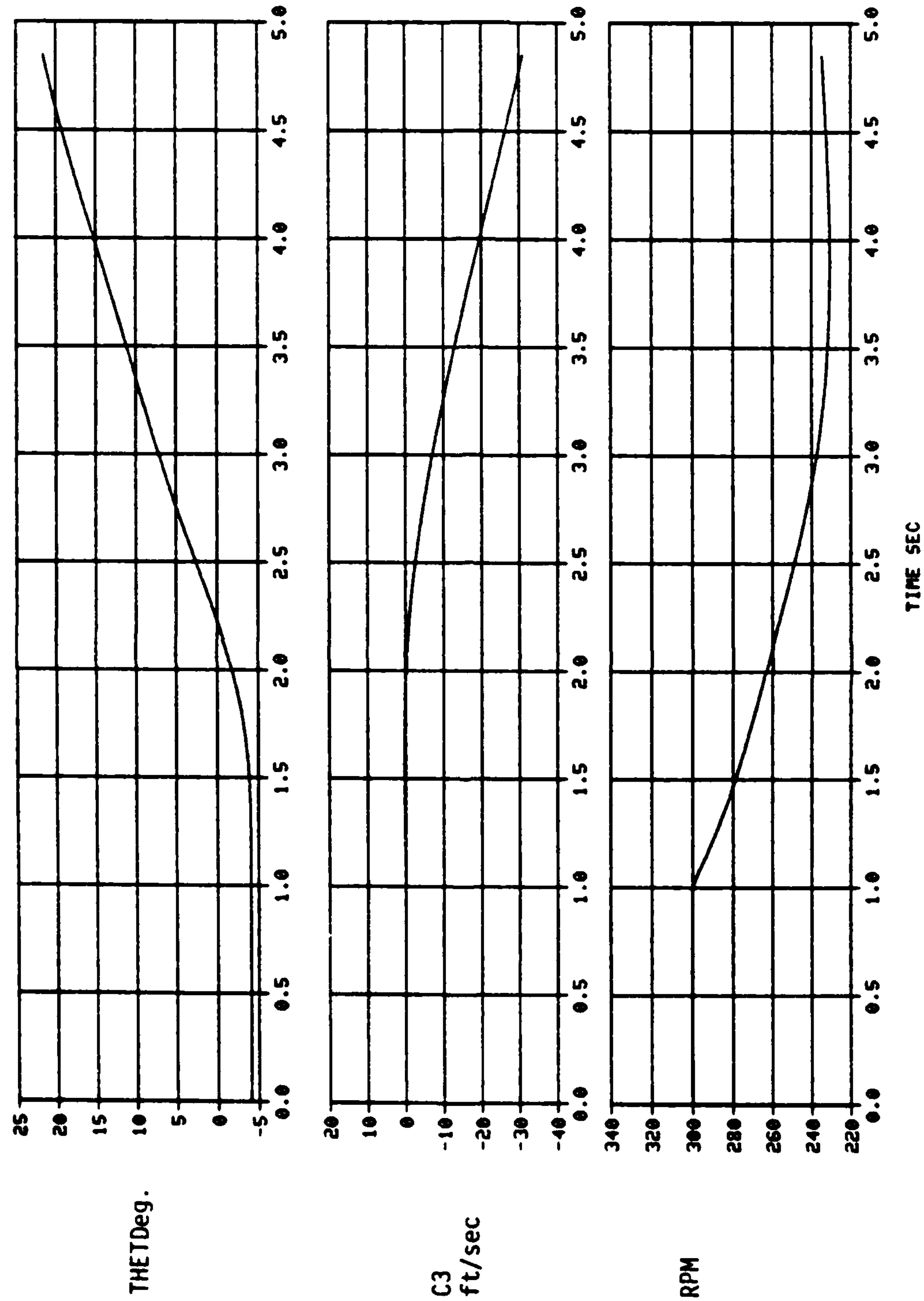

Figure 34. Simulated Engine Failure with Controls Fixed, V = 120 knots

three percent loss in speed shown in reference (f). The most probably explanation for this discrepancy is a more gradual loss of engine torque in the actual helicopter than in the simulation.

The simulated maneuvers may be overly demanding in that the specification requires only that the collective controls should be fixed. Appropriate cyclic and pedal control inputs would tend to minimize the attitude excursions.

A slightly less severe failure situation was represented by engaging the attitude hold system and leaving the collective stick fixed. The engine failure was simulated as in the previous discussion starting from level flight at speeds of 0, 60 and 120 knots. Attitude excursions resulting from the maneuver are illustrated in figures 35-37. All attitude excursions are well within allowable limits except for heading change in hover which exceeds the allowable 20 degree limit. Actual pilot control strategy is likely to be somewhat between the two extreme control laws used in the simulation. Lightly damped attitude oscillations occur primarily because the model control system gains are not properly optimized.

12. Turbulence Response in Hover

The model was trimmed in hover with the position, heading, and altitude hold controllers engaged. Altitude and heading gains were held at nominal values currently used on the aircraft. Position and attitude hold gains were selected based on the turn over a spot performance. Turbulence was represented by adding a sinusoidal contribution to each of the three steady wind components. The magnitude of each component was set to five feet/sec. and the frequency was selected to assure a realistic disturbance.

The following gust equations were used:

$$U_{gust} = 5.0 \times \text{sine } (3.0 \times \text{time}) \text{ - ft/sec}$$

$$V_{gust} = 5.0 \times \text{sine } (4.0 \times \text{time} + 1.57) \text{ - ft/sec}$$

$$W_{gust} = 5.0 \times \text{sine } (5.0 \times \text{time} + 4.7) \text{ - ft/sec}$$

Figure 38 shows the response of the model to the simultaneous application of five foot/second sinusoidal gusts in all three axes. All variables show a sinewave like oscillation in response to the simulated gusts. The collective control moves approximately ± 0.2 inch. Collective motion just meets the recommended maximum excursion and the altitude excursion is well within the suggested ± one-foot limit. The control gain implemented in this analysis is probably too high for gusty air. In addition, the control model does not consider any noise or delays in the feedback sensors which would deteriorate the performance. Lateral stick and rudder pedals show very little disturbance. Longitudinal stick shows more activity, but it is still within the recommended limit of ± one-inch. Bank and yaw attitude excursions are very small. Pitch attitude deviation is reasonably small but could be reduced by increasing the attitude gain at the price of larger control movement.

Overall the aircraft model response to turbulence appears reasonably mild and acceptable. Additional analysis is required to determine the effects of ship airwake turbulence which can be much more severe than the case examined here. Altitude and pitch control show the greatest response to the turbulence. Thus, the collective and longitudinal controls may be the limiting factors in flight through severe turbulence. Pilot comments from the NASA Ames moving SH-2F simulation indicates excessive response to ship airwake turbulence. Additional study is needed to determine the cause of the unrealistic simulator characteristics.

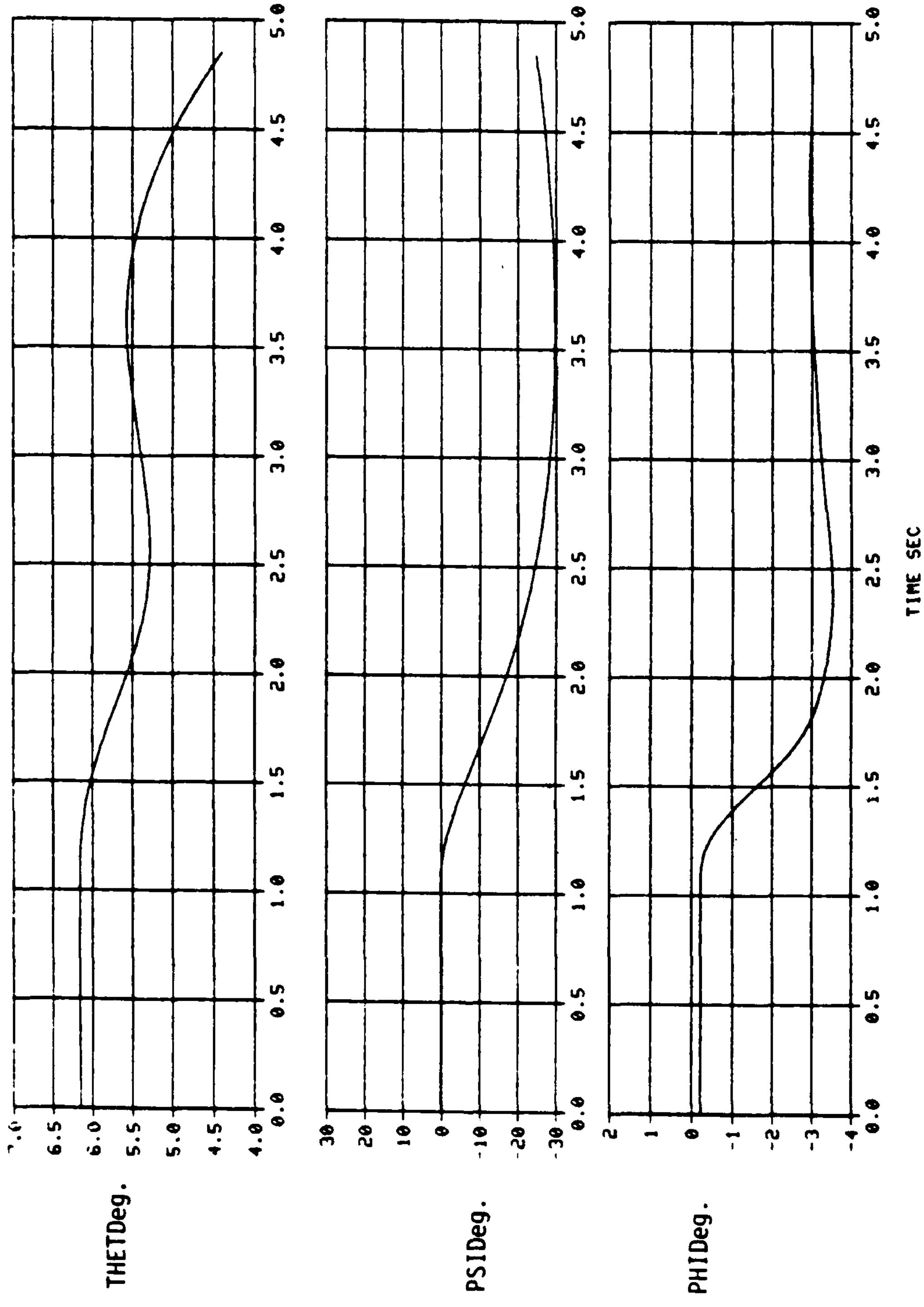

Figure 35. SH-2F Simulated Engine Failure with Fixed Collective, V = 0

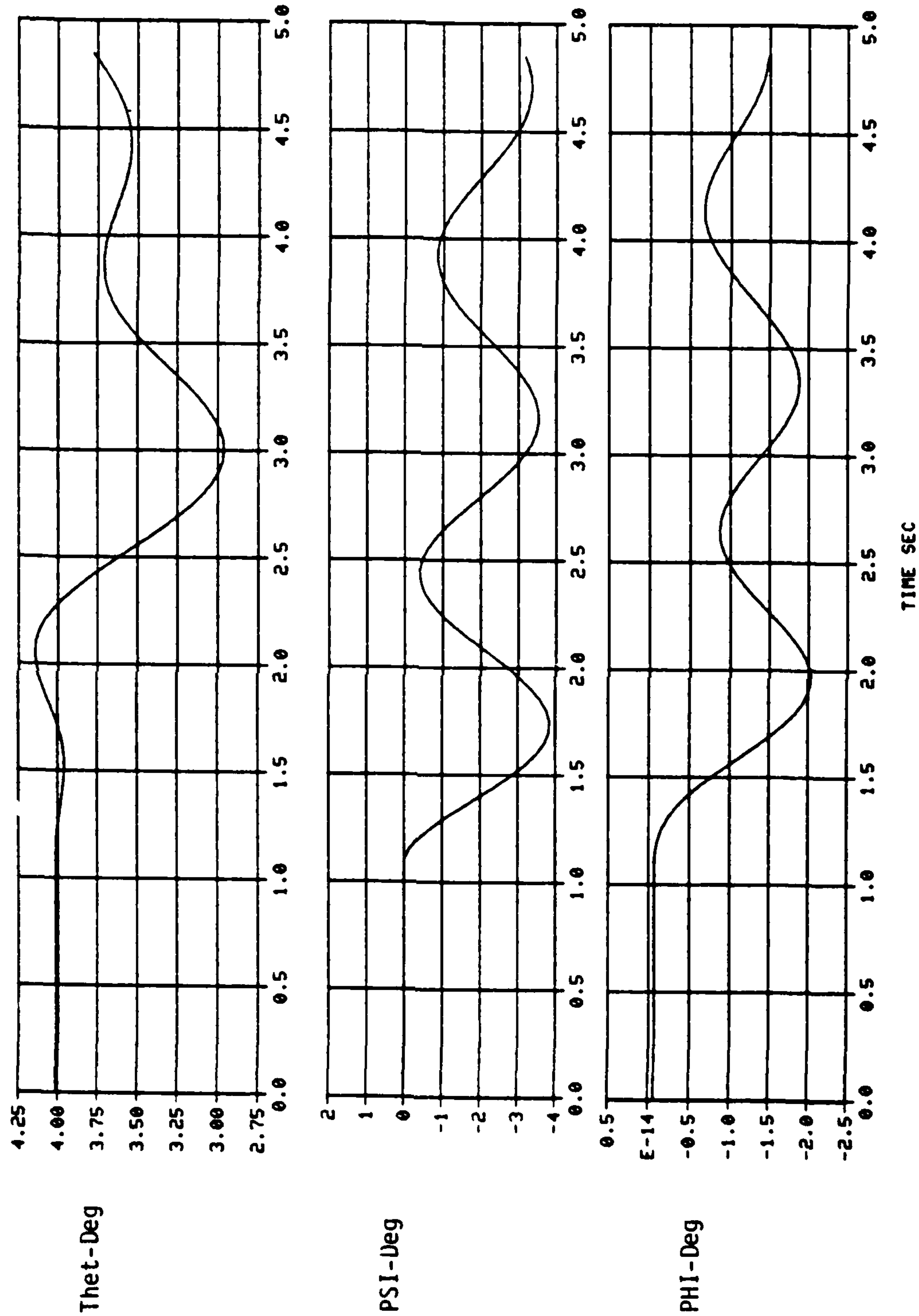

Figure 36. SH-2F Simulated Engine Failure, Collective Fixed, V = 60 knots

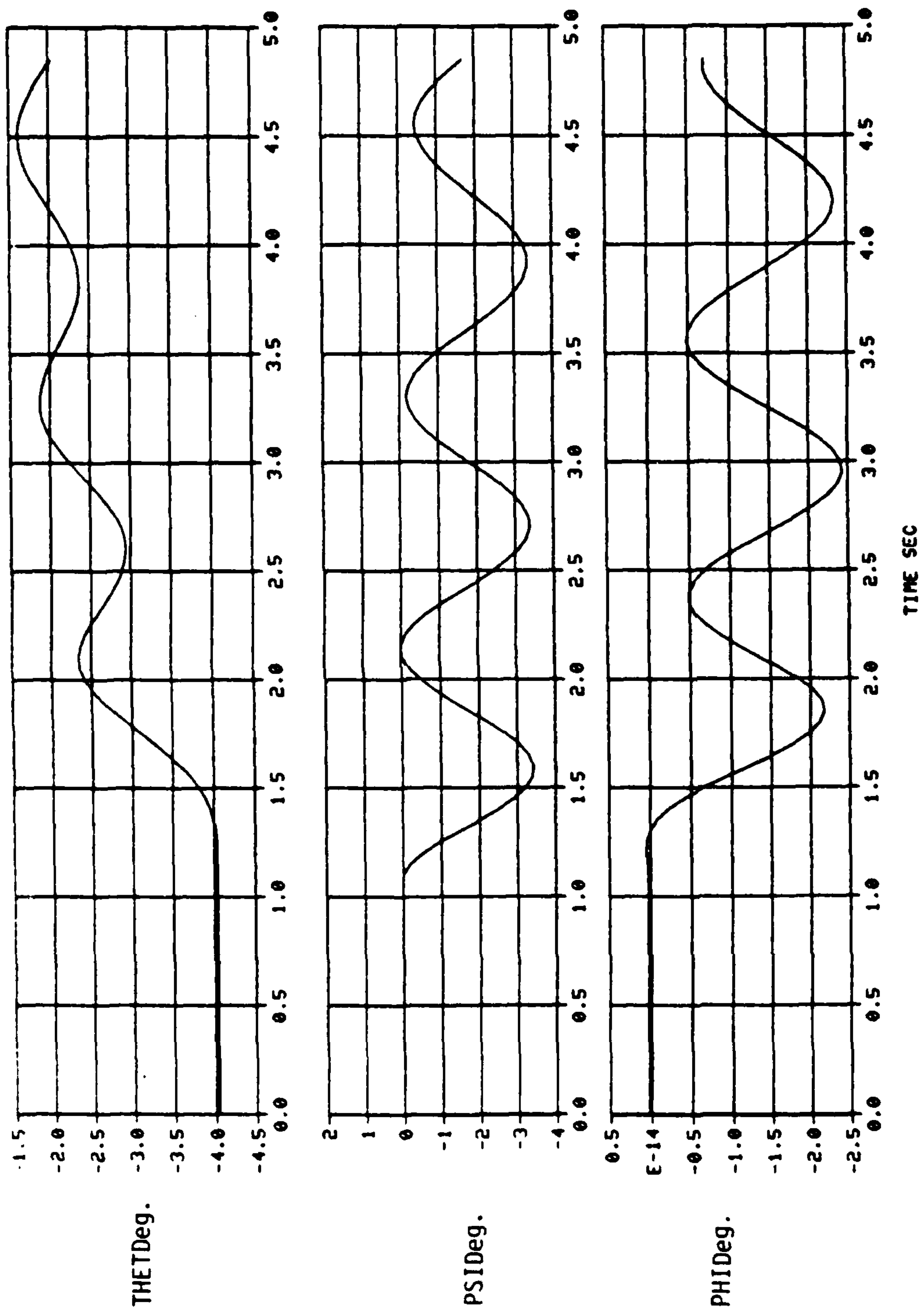

Figure 37. SH-2F Simulated Engine Failure, V = 120 knots, Fixed Collective

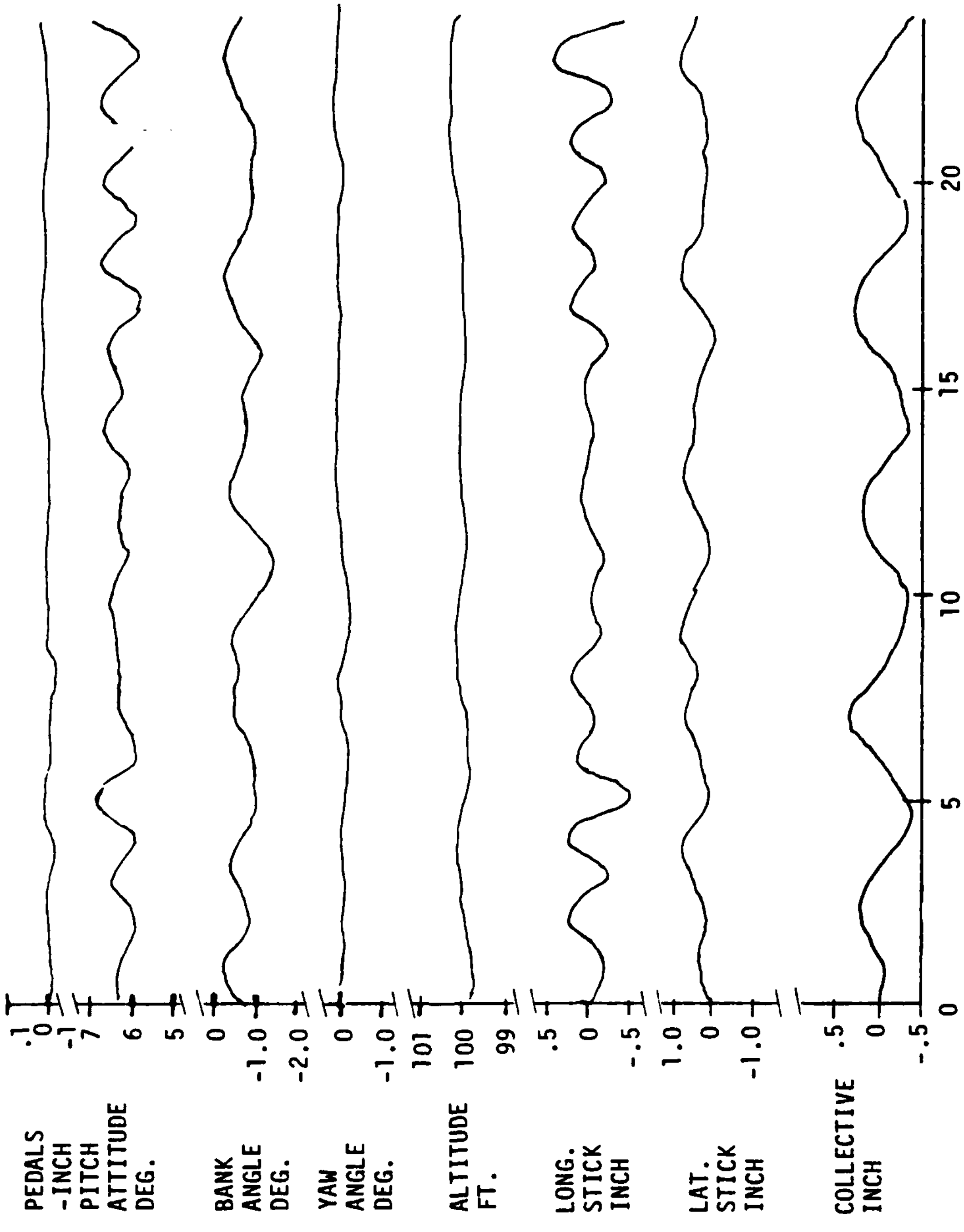

Figure 38. SH-2F Model Gust Response with Position and Altitude Hold

CONCLUSIONS

The SH-2F batch simulation model proved to be a useful tool for preliminary evaluation of the ease of application of MIL-H-8501A. A number of apparent limitations or deficiencies in the specifications were uncovered. However, the analysis was limited by a lack of pilot opinion data needed to evaluate the numerical portions of the specification.

The SH-2F Model satisfies nearly all of the requirements of MIL-H-8501A suggesting that the model should be reasonably satisfactory. Longitudinal control power and dynamic stability satisfy all requirements. Bank response of the model was somewhat sluggish compared with available flight data near hover. Model yaw rate response to rudder was also significantly slower than comparable flight data. However, the slow roll and yaw response resulted from modifications made to the model based on test pilot criticism of the original model dynamics. The use of test pilot comments to tune the dynamics of a ground based simulator is not a reliable method for assuring dynamic accuracy of the simulator. Careful matching of the model dynamic characteristics with flight test data must be performed first. Pilot comments are needed for final simulator acceptance tests, but in general they cannot be used to refine the model unless the pilot objections can be definitely correlated with measured differences between the flight and simulator dynamics.

Both the SH-2F helicopter and the simulator model exhibit excessive collective to pitch coupling. This coupling increases with forward speed and causes a significant deterioration in autorotation characteristics.

Experience developed in working with the helicopter flying qualities specification suggested a number of areas where improvements are needed. The quick stop requirement should specify a minimum allowable altitude excursion for the helicopter. Based on limited test data, a declaration rate of five ft./sec^2 is suggested. Altitude excursion probably should be limited to ± 50 - ± 100 feet.

The static speed stability is difficult to demonstrate inflight because of shallow gradients and unstable dynamics found in most helicopters. In addition, it appears to be a weak constraint on dynamic stability. It may be desirable to require a stable stick gradient established for a range of level flight trim speeds. If the present form of the longitudinal speed stability test is retained, the requirement should be reworded to account for the engine governor rather than a manual throttle.

Dynamic stability requirements should limit aperiodic divergence as well as oscillatory instability. The minimum time to double amplitude should be limited regardless how long the period of the motion might be.

It would be highly desirable to provide an alternate form of the attitude response criteria in terms of available control power acceleration. A criteria in this form could be more easily used in preliminary design studies than the current attitude response criteria. However, the attitude response criteria is most suitable for flight test evaluation because angular acceleration can not be easily measured. The roll response criteria should be changed from one-half second to one second. It is nearly impossible to measure response accurately over a one-half second period. The bank requirement should be transformed to be consistent with the pitch and yaw criteria.

The concave downward requirement does not significantly restrict dynamic stability. This requirement appears to be a carry over from very early helicopter designs when stability was virtually non-existent. It could probably be deleted without significant loss if the dynamic stability requirement were updated and strictly enforced.

Evaluation of simulated autorotation response shows that it may be difficult to comply with the required two-second delay in collective control application following engine failure. Assuming a reasonable level of pilot skill and attention, it may be satisfactory to reduce the required delay to one second. However, piloted simulation studies are needed to establish minimun response times.

LIST OF REFERENCES

a. Helicopter Flying and Ground Handling Qualities; General Requirements for, MIL-H-8501A, 3 April 1962.

b. Trainer, Light Airborne Multi-Purpose Systems (Lamps); SH-2F Helicopter Weapon System Device 2F106, Mathematical Model Report; 7 September 1977; Reflectone, Inc.

c. Results of the Handling Qualities Demonstration for the Model AH-1T Helicopter, Report No. 209-099-436, Bell Helicopter Company, 5 May 1977.

d. CH-53E Handling Qualities Data, Addendum B, MIL-D-2322A (AS), CH-53E Type Specification.

e. A Compilation and Analysis of Helicopter Handling Qualities Data, Volume One: Data Compilation - Systems Technology Report No. 1087-1, March 1979.

f. Flight Fidelity Evaluation of the SH-2F Weapons System Trainer - LT C. Woomer and Mr. G. Carico, Naval Airtest Center, 31 March 1978.

g. Phase II Navy Preliminary Evaulation of the Model HH-2D/101 Rotor System LCDR R. Parkinson, et al., FT-71R-72, Naval Air Test Center, 19 October 1972.

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